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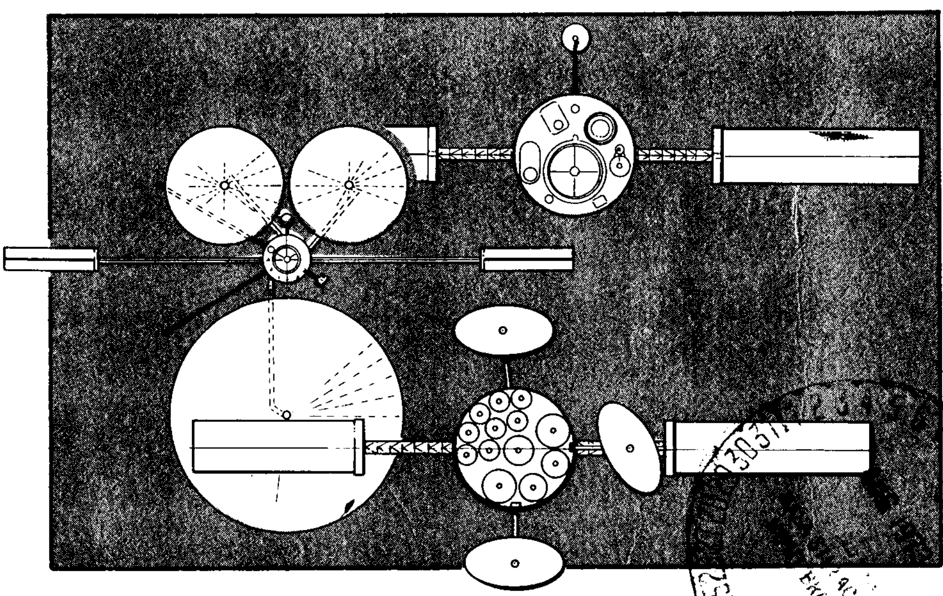
GEOSYNCHRONOUS PLATFORM DEFINITION STUDY

Volume V

GEOSYNCHRONOUS PLATFORM SYNTHESIS



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Volume V
GEOSYNCHRONOUS PLATFORM SYNTHESIS

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FOREWORD

The Geosynchronous Platform Definition Study was a pre-Phase A analysis conducted by the Space Division of Rockwell International Corporation (Rockwell) under Contract NAS9-12909 for the Lyndon B. Johnson Space Center of the National Aeronautics and Space Administration. The study explores the scope of geosynchronous traffic, the needs and benefits of multifunction space platforms, transportation system interfaces, and the definition of representative platform conceptual designs. The work was administered under the technical direction of Mr. David Brown (Telephone 713-483-6321) of the Program Planning Office/Future Programs Division of the Lyndon B. Johnson Space Center.

This report consists of the following seven volumes:

Volume I - Executive Summary	SD 73-SA-0036-1
Volume II - Overall Study Summary	SD 73-SA-0036-2
Volume III - Geosynchronous Mission Characteristics	SD 73-SA-0036-3
Volume IV, Part 1 - Traffic Analysis and System Requirements for the Baseline Traffic Model	SD 73-SA-0036-4 Part 1
Volume IV, Part 2 - Traffic Analysis and System Requirements for the New Traffic Model	SD 73-SA-0036-4 Part 2
Volume V - Geosynchronous Platform Synthesis	SD 73-SA-0036-5
Volume VI - Geosynchronous Program Evaluation and Recommendations	SD 73-SA-0036-6
Volume VII - Geosynchronous Transportation Requirements	SD 73-SA-0036-7

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ABBREVIATIONS

ASCS	Attitude stabilization and control system
ATS	Applications Technology Satellite
CCD	Charge coupled device
CCIR	Consultative Committee for International Radio
CM	Crew module
C/N	Carrier-to-noise ratio
COMM	Communications
Comsat	Communications Satellite
CSM	Common support module
DMS	Data management subsystem
Domsat	Domestic Communications Satellite
ECS	Environmental control subsystem
EIRP	Effective isotropic radiated power
EPS	Electrical power subsystem
FDMA	Frequency division multiplexing
FM	Frequency modulation
GEOPAUSE	Geodetic satellite in polar geosynchronous orbit
Geoseps	Geosynchronous solar electric propulsion stage
Intelsat	International Communication Satellite
IPACS	Integrated power and attitude control system
Mersat	Metrology and Earth Observations Satellite
Navsat	Navigation and Traffic Control Satellite

OTS	Orbital transportation system
PCM	Pulse code modulation
PSK	Phase shift keying
RCS	Reaction control subsystem
RSU	Remote service unit
SATA	Small Application Technology Satellite
SEP	Solar electric propulsion
SGLS	Space-ground link subsystem (part of U.S. Air Force Satellite Control Facility)
SNR	Signal-to-noise ratio
SSM	Spares storage module
STDN	Spaceflight tracking and data network
STS	Space transportation system
TDMA	Time division multiple access
TDRS	Tracking and Data Relay Satellite
TPS	Thermal protection subsystem
TT&C	Tracking, telemetry and command
UHF	Ultra high frequency
VHF	Very high frequency
WARC	World Administrative Radio Conference
XMTR	Transmitter

1.0 INTRODUCTION

This volume, Geosynchronous Platform Synthesis, presents the development of the platform configurations, support subsystems, mission equipment, and servicing concepts. The complement of equipment is synthesized in accordance with the operational and system requirements derived in Volume IV. A common support module is developed; subsystem concepts are traded off; data relay, TDRS, earth observational, astro-physics, and advanced navigation and traffic control mission equipment concepts are postulated; and ancillary equipment required for delivery and on-orbit servicing interfaces with geosynchronous platforms is grossly defined. The general approach was to develop a platform concept capable of evolving through three on-orbit servicing modes: remote, EVA, and shirtsleeve.

The definition of the equipment is to the assembly level. Weight, power, and volumetric data are compiled for all the platforms. The programmatic comparisons and costing data presented in Volume VI are based upon the servicing concepts, platform configurations, and equipment definitions presented in this volume.

2.0 SUMMARY

The data pertaining to the synthesis of geosynchronous platforms are presented in three major sections. In Section 3.0, Support Subsystems Synthesis, the common structure and subsystem concepts are derived. Ancillary servicing equipment and typical interfaces are also defined. The mission equipment for all types of data relay platforms is defined in Section 4.0. Earth observations and astro-physics discipline mission equipment are defined in Section 5.0, Observation Platform Synthesis.

SUPPORT SUBSYSTEMS SYNTHESIS

Various configurations are evaluated for volumetric efficiency, accessibility, and on-orbit servicing adaptability. The overriding consideration in the selection of a toroidal structure for the common module is the adaptability from auto-remote to manned servicing (including shirtsleeve) operations. The toroid is slightly less efficient volumetrically than cubic or rectangular solid concepts, but to incorporate or convert the latter configuration to a shirtsleeve servicing mode resulted in volumetric, weight, or operations contraints.

The toroidal concept could be conveniently configured to accommodate 12 replaceable modules of a standard 24-inch by 20-inch by 24-inch size. Therefore, a baseline requirement was established that all assemblies, including mission equipment assemblies, be packaged in this standard module. Upon completion of the synthesis process, only the K_{HI}-band 84-channel, 15-antenna transponder assembly exceeded the standard dimensions. This particular assembly is approximately 24 inches by 20 inches by 36 inches. A maximum of two of these assemblies is required on any single platform. By installing these assemblies in alternate compartments in the toroid, adequate clearances for modular replacement can be maintained.

The rationale for standardized packaging of assemblies is to simplify the servicing operation. Admittedly, some of the equipment does not require the entire modular volume, but with a standard size only a single attach/detach and manipulator end effector design is required. Also, the storage facility on the servicing unit can be standardized. Any replaceable module can be stored in any of the compartments. Neither two manipulators nor a temporary holding port is required for the modular interchange operation. In essence, if the servicing unit has one empty compartment at mission initiation, the interchange operation can be accomplished in any sequence desirable, and can even be changed in real time, either by ground control (auto-remote servicing) or by astronauts (manned servicing).

Trade studies were conducted to determine preferred subsystem concepts for the support functions. In general, the most stringent performance requirements of the satellites or platform for each function were the drivers in the synthesis procedure. The two exceptions were the attitude stabilization and control system (ASCS) and the thermal protection system (TPS) concepts.

Some of the observational types of satellites and platforms require pointing accuracies of less than one arc second and stabilization of less than an arc second per second. These are unrealistic requirements for a centralized ASCS. A pointing accuracy of 10 arc seconds and a stability of 1 arc second per second are more practical centralized ASCS capabilities. The common approach in achieving tighter control is either to use the sensor that requires the precision accuracy as an integral part of the control loop or to separately gimbal the sensor in a cascaded control loop.

A single TPS concept for the assemblies of the support system is feasible, but special provisions are required to accommodate the power dissipation requirements of some of the mission equipment modules. Restoration of an adequate thermal conduction path between a replacement module, which has a power dissipation greater than 100 watts, and its coldplate by only a structural interface is questionable. Inclusion of a crushable thermal grease packet on the mating surface of this class of replaceable module is recommended. In addition, the total power dissipation requirement of the mission equipment for the different types of platforms varies significantly. The space radiators for the mission equipment complements were individually sized to meet the requirements of each platform.

The most stringent support subsystem requirements for the platform are listed in Table 2.0-1. Table 2.0-2 summarizes the selected centralized support subsystem concepts. All of the support subsystem concepts are compatible with at least "fail-degraded" reliability criteria.

Table 2.0-1. Most Stringent Support Subsystem Requirements

Support Subsystem	Requirement	Platform Driver
Electrical power	2.0 kilowatts	Earth observations
Pointing	0.1 arc seconds	Astro-physics
Stability	0.05 arc seconds per second	Astro-physics
Data handling	50 Mbps	Earth observation
Impulse/propulsion	19000 lb-sec/88 lb hydrazine	Astro-physics
Module power dissipation	500 watts	Earth observations and astro-physics

Table 2.0-2. Selected Support Subsystem Concepts

Support Subsystem	Concept	Capability
Electrical power	Solar array-battery	2 kilowatts continuous for 10 years
Pointing	Charge coupled device star tracker	10 arc seconds[1]
Stabilization	Three orthogonal reaction wheels	1 arc second/second[1]
Data handling	High-gain antenna (TM)	50 Mbps
	Omni antenna Command TM	 64 bps 1 Kbps
Impulse propulsion	Four independent modules; four 1-pound thrust engines/module; hydrazine	39,000-lb-sec impulse; 10-year propellant supply (2.5-year supply for N-S stationkeeping)
Thermal protection system	Module coldplates; heat pipe thermal transport; space radiators; grease packet required for modules with 100-watt power dissipation.	Controls between 40°F and 70°F

[1]Provisions included for direct input from sensors requiring more stringent control accuracies.

Some of the platforms require significantly less power than 2000 watts. Therefore, power system concepts for 500, 1000, and 1500 watts were also synthesized because this particular subsystem is adaptable to incremented performance levels.

Although only two RCS modules are required to meet the impulse requirements, two additional modules are included to provide limited north-south stationkeeping capability for data relay platforms and to increase the reliability of the system. The toroid structural concept can accommodate 12 standard modules. Only eight are required for the other support system assemblies. Therefore, increasing the RCS modules from two to four units does not impact the basic structural design.

ON-ORBIT SERVICING CONSIDERATIONS

Servicing equipment concepts for both auto-remote and manned servicing operations were derived. The gross weight of the auto-remote and manned concepts are 1650 pounds and 6050 pounds, respectively, if two equipment storage rings are provided in each case.

Concepts for the physical attachment of the logistics system to the platform for both unmanned and manned servicing modes are defined. For the auto-remote servicing mode, a hinge mechanism is defined which permits the pivoting of the platform, after docking, perpendicular to the centerline of the tug. This operation permits internal access to the platform by the service unit manipulator for replaceable module interchange. A concentric dual docking ring adapter, which provides the transition from the basic opening of the platform to the standard space shuttle docking port, is also defined.

Electrical and mechanical interfaces between a replaceable module and the platform structure are defined. A remotely activated docking mechanism concept for module retention that is adaptable for both unmanned and manned servicing modes is identified. An electrical connector similar to that used on current outer planet mission spacecraft is proposed.

DATA RELAY PLATFORMS

Four configurations (one for each global region) are developed for the platform equivalents of the Domsat, Comsat, and traffic control satellites of the baseline traffic model. The only differences between the platforms are the antenna and transponder assembly complements.

Concepts other than the toroid were evaluated for housing the mission equipment. Only a small percentage of the compartments of the standard toroid equipment ring is required to house the mission equipment of some of the platforms. Some configurations result in less weight, but in accommodating the three servicing modes (auto-remote, EVA and shirt-sleeve), the concepts evolved essentially into the basic toroid shape. Rather than develop multiple structural configurations, it was decided that it would be more economical to utilize the basic toroid structure for the mission equipment and pay the slight weight penalty.

Table 2.0-3 summarizes the characteristics of the platforms that are the equivalent of the Domsat, Comsat, and traffic control satellites of the baseline traffic model. Two of each type are required in each of the four global regions. The L-band equipment is for the relay of data for traffic control purposes. The C- and K-band equipment is for the Domsat and Comsat functions.

Table 2.0-3. Baseline Traffic Model Data Relay Platforms

Mission Equipment (per Platform)	Transponders				Antennas				
	C	K_{HI}	K_{LO}	L	C	K_{HI}	K_{LO}	L	Shaded
REGION I	1			1	3			1	
REGION II	2			1	4		1	1	1
REGION III	1	1		1	4	2		1	
REGION IV	2	1	1	1	2	2	2	1	3

Platform Summary	REGION I	REGION II	REGION III	REGION IV
Total weight (lb)	2759	2986	3274	3835
Total power (watts)	818	1418	1298	1723

The tracking and data relay platforms are identical for both the baseline and new traffic models. They consist of repackaging of the equipment of the TDRS satellite into standardized replaceable modules. The characteristics of the platforms are as follows:

Low data rate relay: 10 kbps simultaneously from 20 users
Medium/high data relay: 50 Mbps simultaneously to/from two users
Command uplink: 10 kbps to one user at a time
Total weight: 2651 pounds
Total power: 491 watts

The data relay platform equivalents of the Domsat and Intelsat satellites of the new traffic model consist primarily of different complements of the mission equipment of the same types of data relay platforms for the baseline traffic model except for the L-band equipment. The L-band equipment, which was associated with the traffic control function that was defined in the baseline model, is not required. The definition of this function in the new traffic model imposes unique orbit placement requirements that are incompatible with Domsat and Intelsat placement requirements.

The Intelsat and Domsat functions are on separate platforms in the platform equivalent of the new traffic model. The separation is not a result of incompatibility of functions. Rather, the projected data traffic dictates multiple platforms for each global region. From a mechanization standpoint, it is more logical to utilize the full frequency spectrum on each platform for either Domsat or Comsat rather than attempt to intermix the two. The selected concept reflects the minimum number of platforms that will meet the requirements of the traffic model.

Table 2.0-4 summarizes the data relay capacity of each type of platform. Table 2.0-5 summarizes the platform characterisitcs. Minor variations in the total weight (± 100 pounds) between platforms for the various regions exist because of different antenna complements.

Table 2.0-4. Mission Equipment Capacity

Transponders \ Platform Type	Intelsat				Domsat			
	Regions				Regions			
	I	II	III	IV	I	II	III	IV
C-band (24 channels)	1	1	1	1	1	1	1	1
K_{LO}-band (24 channels)					1	1	1	1
K_{HI}-band (84 channels)	2	2	2	2	2	2	2	2

Table 2.0-5. New Traffic Model Data Relay Platform

Characteristic \ Platform Type	Intelsat	Domsat
Total weight (lb)	3600	1478
Total power (watts)	4000	1603

The navigational and traffic control platforms for the new traffic model are separate because of new requirements for monitoring polar in addition to lower-latitude traffic and for navigation data to ships and aircraft in addition to basic traffic control information. The functional capability of the platforms is the same as that of the satellites in the new traffic model. The resultant platform characteristics are:

Total weight, 2799 pounds
Total power, 1843 watts

In all cases, the data relay platform configurations use two standard equipment rings--one for the support or subsystem equipment and one for the mission equipment. The number and frequency bands employed vary from platform to platform, as does the antenna complement; but in general, a single development of L-, C-, K_{HI}-, and K_{LO}-band transponders and antennas will permit the synthesis of the mission equipment for all of the data relay platforms.

OBSERVATIONAL PLATFORMS

A synthesis of representative earth observation and astro-physics mission equipment is presented. Because of inadequacies in the definition of applicable geosynchronous sensors and associated equipment, low-orbit satellites were evaluated to determine the desirability of their functions being performed from geosynchronous orbit. Where applicable, and when advantages for operations at geosynchronous altitudes were identified, the equipment was scaled to provide the equivalent capability from geosynchronous altitude.

The synthesis technique resulted in the identification of 22 sets of observational equipment. A combining process was conducted to minimize duplication of equipment and ensure both operational and equipment compatibility. Five platforms are defined for geosynchronous orbit. They are:

- Earth observations
- Solar astronomy
- Stellar astronomy
- Plasma physics
- High-energy physics

The characteristics of each platform are summarized in Table 2.0-6. Four earth observation platforms are required to provide global coverage for both meteorological and earth resource observations. The astro-physics platforms are considered to be only of U.S. origin in the baseline traffic model. In the new traffic model, foreign astro-physics space elements are also indicated. Because of the lack of definition of these foreign space elements, for the purposes of this study they are considered to consist of the same mission equipment groups as the U.S. astro-physics platforms.

Table 2.0-6. Observational Platform Characteristics

Platform Summary	Weight (pounds)	Power (watts)
Earth observations (one per region)	8496	1634
Solar astronomy	8302	1328
Stellar astronomy	5896	630
Plasma physics	4102	1280
High-energy physics	8499	580

3.0 SUPPORT SUBSYSTEMS SYNTHESIS

It is a well-recognized fact that one of the primary cost factors in any space program is the customized design approach in the development of each space element. The objective of this study is to determine the feasibility of development of geosynchronous platforms. Two factors of that objective are (1) to establish a configuration that will permit the reduction in total end items in geosynchronous orbit, and (2) to extend the usable life of mission equipment by conducting on-orbiting servicing. These two approaches could significantly reduce total programs costs but the potential of customized platforms is still apparent. Thus a major emphasis task throughout this study is to not only develop platform concepts but also maximize commonality of equipment and operations between the various types of platforms.

In this section the analyses that were conducted to achieve platform commonality are presented. The spectrum of servicing modes range from auto-remote to shirtsleeve operations. In order to facilitate an evolutionary transition from one mode to the other, it was necessary to define the design drivers of each mode and synthesize a configuration that would accommodate all projected servicing modes. The prime driver in the case of auto-remote servicing was the opening in the platform for manipulator articulation during module interchange. Shirtsleeve servicing was the major driver in the platform configuration selection. The platform must be capable of being pressurized and adequate work space must be provided for the crewman during maintenance operations in the platform. Section 3.1 presents the configuration synthesis data.

The usual procedure in the derivation of subsystems is to evaluate the individual space element performance requirements and then optimize the subsystem equipment. In this study the approach was to establish the spectrum of performance requirements and then select a subsystem concept that will meet the most stringent requirements. Admittedly in some cases the support systems are "over designed" for the mission equipment. Where practical, incremental plateaus of equipment are defined. For example, power systems for 500, 1000, 1500, and 2000 watts are identified. A summary of support requirements is presented in Section 3.2. Subsystems trades are contained in Section 3.3.

The commonality principle extends to servicing operations also. The ancillary equipment and space elements that are derived in Section 3.4 reflect a maximizing of commonality for auto-remote, suited, and shirtsleeve servicing modes. The concepts permit transition from auto-remote to shirtsleeve servicing without any significant on-orbit platform modifications. The option also exists to mix manned and auto-remote servicing if desired.

Concepts for both internal and external platform interfaces are presented in Section 3.5. An electrical interconnect for RF as well as signal/power interfaces is defined. Latch/unlatch mechanisms for replaceable module interchanges in any of the servicing modes are identified. Docking and the mechanical interface between servicing space elements and the platform are also defined.

3.1 SUPPORT MODULE OPTIONS

Various concepts of space elements that include on-orbit servicing provisions have been, and are being, studied by the NASA and NASA contractors. The concepts have been for auto-remote or manned servicing and in some cases the concept was optimized for a particular set of mission equipment. The geosynchronous platform configuration for on-orbit servicing has two unique requirements: (1) it must accommodate both auto-remote and manned servicing including shirtsleeve operations, and (2) the concept must accommodate the maximum number of payloads. The derived geosynchronous platform configuration and the rationale for its selection are presented in this section.

COMMON SUPPORT MODULE

One major part of any platform that is a candidate for standardization is the aggregation of support or utility functions. Included in this group are electrical power, data handling, telemetry and command, stabilization and control, attitude determination, and attitude control. The objective is to select a common structure to house the support function assemblies in the various platforms and, if practical, use the same support system modules on all platforms.

Structural Configuration

In the following paragraphs, configurations are examined for both volumetric efficiency and servicing adaptability.

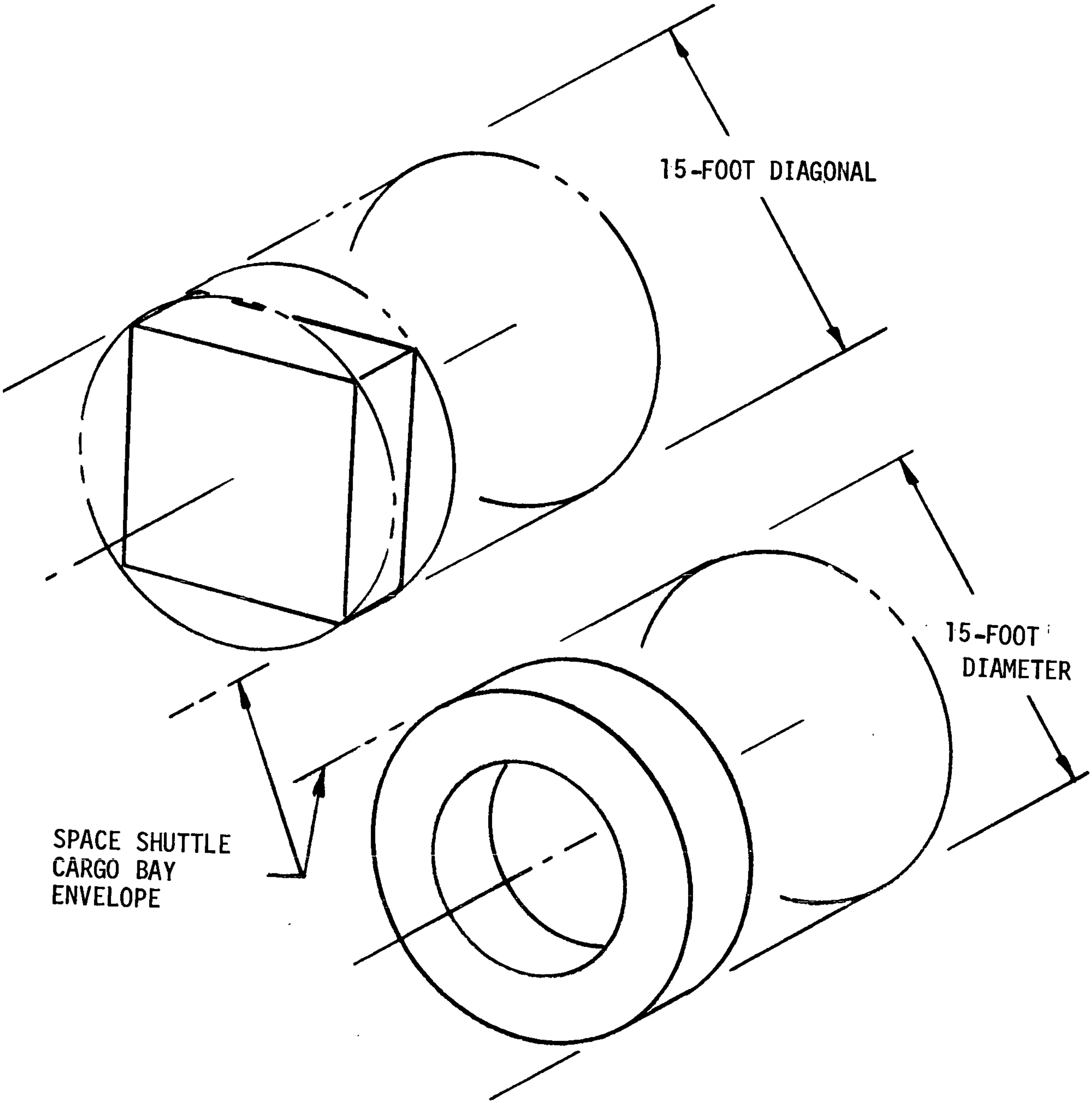
Volumetric Efficiency

The two basic geometric shapes for a common support module are a rectangular solid/cube and a toroid (Figure 3.1-1). The maximum dimension of either shape must be limited to the 15-foot diameter of the space shuttle cargo bay. Therefore, the toroid will provide more usable volume per unit length of structure provided the inside diameter of the toroid does not exceed approximately nine feet.

A more realistic approach would be to include consideration of appendages on the module and at least some clearance in the cargo bay. In the case of the cube, appendages could be located on the sides with ample clearance. For structural clearance the sides could be reduced from 10.5 to 10.0 feet. Reduction of the diameter of the toroid to 12.0 feet would provide ample clearance for both structure and appendages such as reaction jets and possibly sensor pivot mechanisms. An additional consideration is the inclusion of a docking port on the structure. If a 5-foot-diameter docking port is assumed, then the usable volume of the cube is 75 cubic feet per unit length (Figure 3.1-2). The 12-foot-diameter toroid could have an inner diameter of approximately 7 feet and equal the volume per unit length of the cube (Figure 3.1-3.

CUBIC CONFIGURATION

VOLUME PER UNIT LENGTH →	112.5 ft^3



TOROID CONFIGURATION

VOLUME PER UNIT LENGTH =	176 ft^3	
LESS 9-ft DIA. OPENING	63.5 ft^3	→ 112.5 ft^3

Figure 3.1-1. Basic Configurations

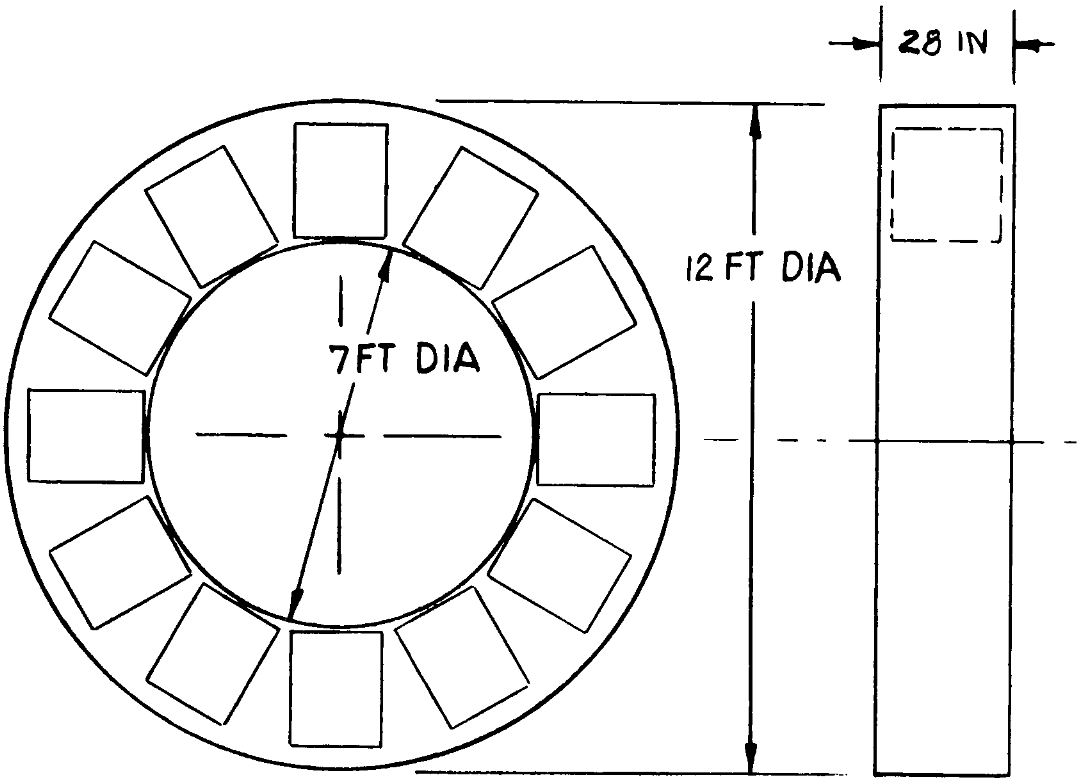


Figure 3.1-2. Standardized Toroidal Configuration

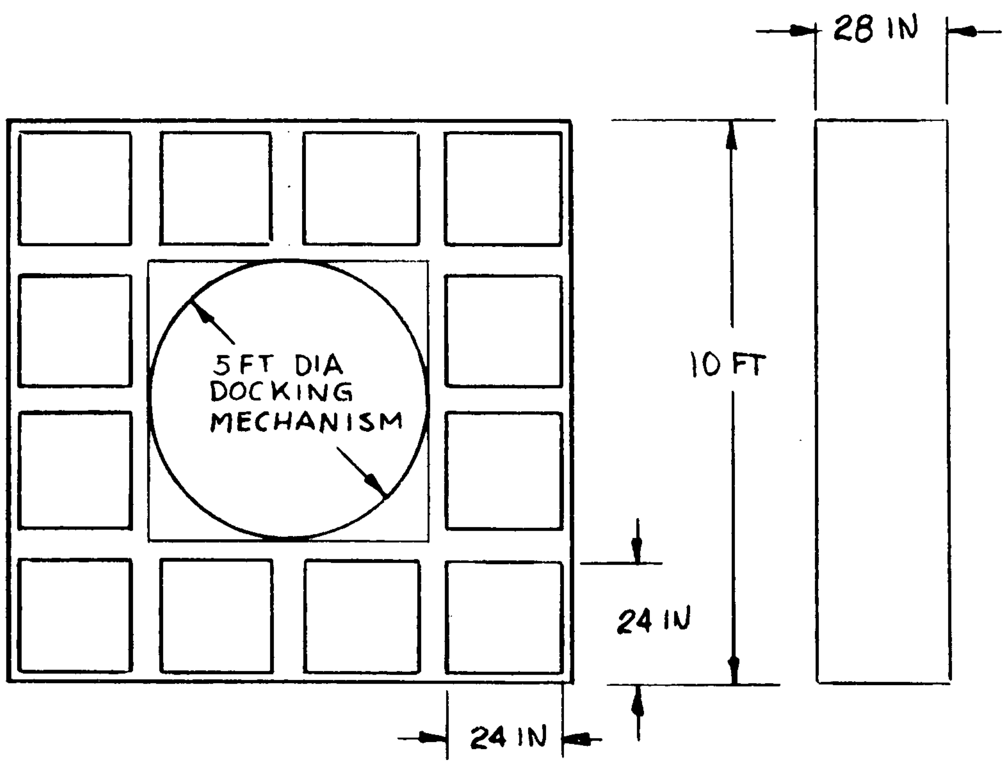


Figure 3.1-3. Standardized Cubic Configuration

Accommodation of modules is of prime importance. A layout of standardized modules is shown in the figures. Both configurations can readily accommodate 12 modules. However, the standard size can be slightly longer in the cubic configuration (24 inches by 24 inches versus 24 inches by 20 inches).

Both configurations can accommodate various sizes of modules; however, the cubic concept offers more flexibility in packaging. A standard replaceable module size is preferred for ease in mechanization of auto-remote servicing and storage of modules. That is, if only a single modular size is employed, then the logistics element requires only one articulation device and one "empty" storage compartment at mission initiation.

From purely a volumetric efficiency and packaging accommodate standpoint, there are no significant advantages or disadvantages between the cubic and toroid configurations.

Servicing Adaptability

The adaptability of the basic configurations for on-orbit servicing must be evaluated for all three modes: auto-remote, pressure-suited, and shirtsleeve. Both the cubic and toroid concepts are adaptable to auto-remote and pressure suit servicing operations. A technique applicable to either configuration for the servicing modes is illustrated in Figure 3.1-4.

Shirtsleeve operations impose unique design requirements. The only practical configuration for a pressurizable volume is cylindrical in shape. Module interchange must be conducted within the pressurized volume. In the case of the toroid, only a pressure hull is required and the modules are changed out from the interior instead of externally. The cubic structure would require a completely new and additional outer pressure structure. In fact, it would become essentially the same as the toroid except the replaceable modules would be orthogonally located rather than radially mounted (Figure 3.1-5). In order to be comparable to the toroid the outside diameter (diagonal of the square) of the pressurizable cubic configuration would be reduced to 12 feet. For inside servicing, the four center modules that are dotted in would have to be removed. Also, the four corner modules would require removal of an adjacent box to be accessible.

If the manned servicing unit included a work area of hangar area that was also 12 feet in diameter and the pressure seal was at the outer edge of the cylinder (Figure 3.1-6), than all 16 modules could be readily replaced. This concept has two major shortcomings. The configuration limits the number of replaceable standardized modules to 16. Two tiers of equipment are not accessible in a shirtsleeve environment. Secondly, the hangar work area would consume a significant portion of the shuttle bay volume, some of the shuttle payload weight capability, and a relatively large percentage of the tug payload capability.

It appears that the only practical configuration for shirtsleeve servicing operations is a toroid configuration. Multiple tiers of equipment can be assembled. Direct access from a crew module through a docking port can be accomplished. No unique adapters are required. Work space is available within the platform; thus, free space requirements in the crew module are less stringent.

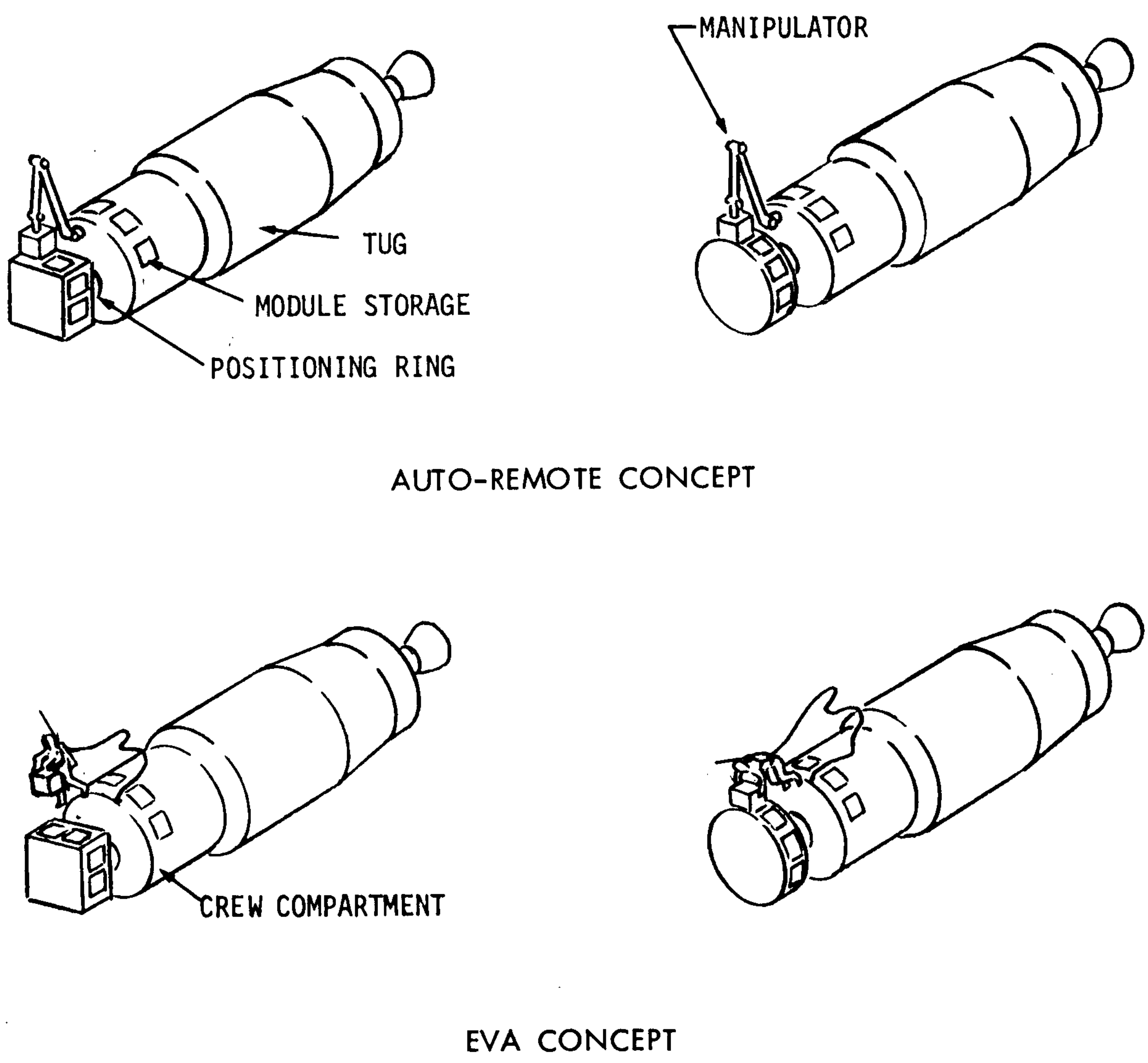


Figure 3.1-4. External Changeout Concept

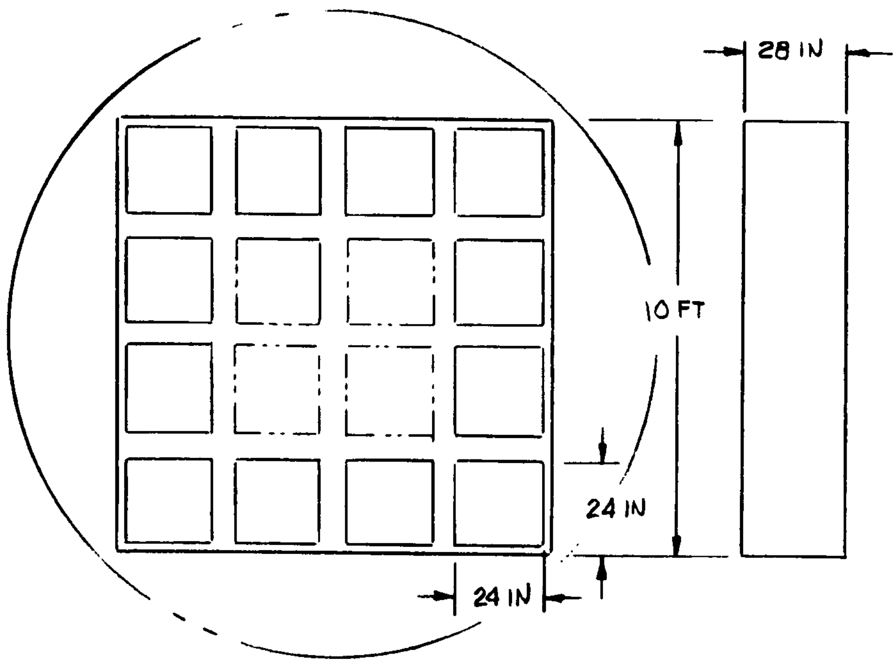


Figure 3.1-5. Pressurizeable Cubic Configuration

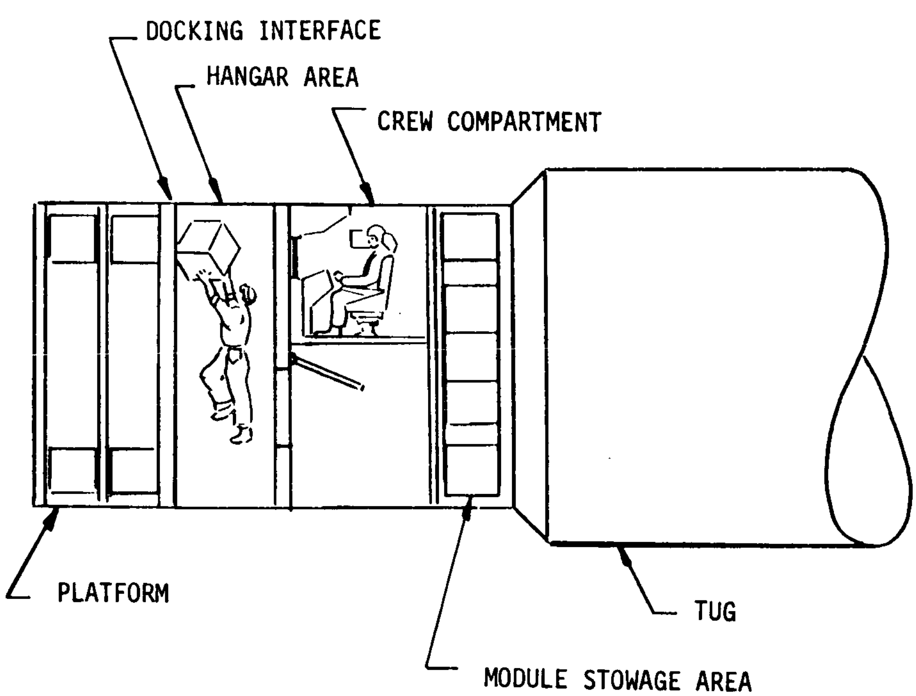


Figure 3.1-6. Hangar Work Area Concept

Evolutionary Concept

One of the most important requirements of the common support module structure is that it will accommodate all three modes of servicing. That is, as the mode of servicing evolves from auto-remote to shirtsleeve the platform design is still valid for on-orbit maintenance operations. The toroid meets these requirements.

Radial mounting of replaceable modules will facilitate interchange operations in the auto-remote mode even with inside changeout. A single articulation mechanism can be used. By proper clocking of the mechanism, comparatively accurate placement of the mechanism with and without a module can be achieved. During the more critical installation and removal operation of modules with the secondary structure, only orthogonal motions would be required.

Pressure-suited servicing operations in a toroid have a distinct safety advantage over all other potential configurations. Inside changeout is again a feasible concept. Thus, the suited crewmen are restrained within a structure. Tethers are not required. The pressure suit can be of an umbilical type rather than be completely self-contained. Crewmen task schedules and task complexity can be significantly increased by the additional freedom of actions that can be permitted in the toroid configuration.

The advantages of the toroid for shirtsleeve servicing operations were pointed out previously.

The geosynchronous platform concept provides an orbital facility that permits replacement of equipment because of failure, wear-out, consumables replenishment, or technology update. The same platform is to be used with either of the three servicing modes. An evolutionary program that entailed the replacement of orbital elements as the servicing modes evolved was considered to be uneconomical. Therefore, the most stringent platform design requirements for each servicing mode were identified and the platform was designed accordingly.

The auto-remote servicing mode presented two unique constraints: (1) the replaceable modules must include a front-mounted latch/unlatch mechanism that can be activated by an articulation device, and (2) adequate clearances in the center area of the toroid must be provided for manipulation of the modules. The design details of the latching mechanism are presented in Section 3.5. It essentially consists of a dual jack screw arrangement, one on each side of the module, that requires minimal torque and force. It was specifically designed to be within the capability of the space shuttle manipulator.

The clearance in the center of the toroid was selected as approximately 7 feet. This permits the centering of the articulation device, extending an arm to a module (3.5 feet) and withdrawing a module 2 feet in depth from its mounting structure. Thus, from the center axis of the toroid to the removed module there is 1.5 feet for the telescoping articulation device. Details of the interchange sequences are presented in Section 3.4.

There were no unique design requirements identified for pressure-suited servicing of the toroid concept. Mobility aids and restraints for both manned servicing modes could be the same.

Shirtsleeve servicing was the primary driver of the configuration. A pressure hull is required and also inside changeout is required. Thermal protection and atmospheric control during shirtsleeve operations is an additional consideration. Surface temperatures must be controlled between 58° and 105 F. This requirement had no significant impact on the basic platform thermal control system. Atmospheric control was not included in the platform design. The infrequency of long durations between manned visits made it totally impractical to provide this function as an integral part of the platform. Atmospheric control must be provided by the logistics element. In all probability, fans and temperature sensors will be required to regulate air circulation between the crew module and the platform.

The 7-foot opening/docking port required for auto-remote servicing will result in the requirement for an adapter between the crew module and the platform. It is assumed that the crew module docking port will be compatible with the space shuttle, which is appreciably smaller than 7 feet in diameter. The details of this docking adapter are presented in Section 3.5. It should be noted that the adapter need not be a penalty on every logistics flight. As the servicing mode evolves to manned servicing the adapter could be brought to the platform on a mission and left there upon departure.

The major differences in the toroid concept for the various servicing modes are illustrated in Figure 3.1-7. The one on-orbit change recommended is the revision of the docking port size by means of an adapter. All other provisions for the various modes of servicing are to be included in the initial platform. It is essentially a shirtsleeve serviceable element upon initial emplacement in orbit.

Support Systems

The support or utility functions that are common to all unmanned space elements are as follows:

- Electrical power
- Stabilization
- Attitude determination
- Attitude control
- Thermal control
- Data processing
- Tracking and telemetry

Development of modular packaging of the assemblies that are required to perform these functions is dependent upon servicing considerations, volumetric requirements, power dissipation, and interface complexity. Components and assemblies of a similar nature should be grouped together. Certain items have a predictable wear-out period. For example, it would be illogical to group solid-state equipment with equipment containing rotating machinery; assemblies containing consumables should not be grouped with wear-out type of equipment, etc. Table 3.1-1 summarizes the preferred equipment groupings and the reasons for the selection.

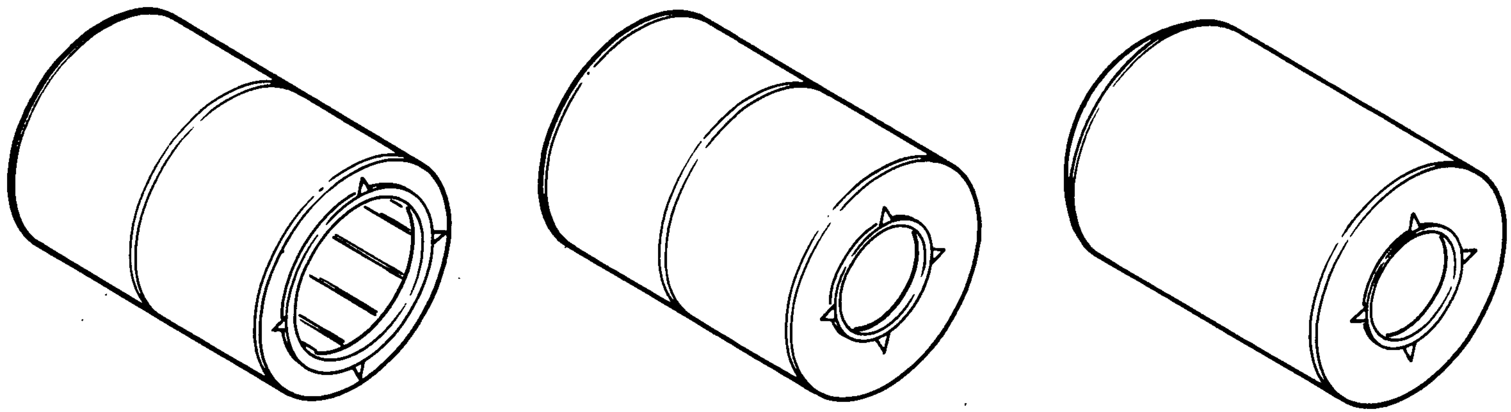


Figure 3.1-7. Servicing Mode Impact on Toroidal Platform

Table 3.1-1. Support System Functional Groupings

Function	Modular Grouping	Rationale
Electrical Power	Battery pack	Failure rate "wear out"
	Solar array and drive mechanism (2)	Wear out Interface complexity
	Power conditioner	Failure rate Technology advances
Stabilization	Reaction wheels	Alignment criticality Wear out Technology advances
	Control electronics	Failure rates Technology advances Interface criticality
Attitude Determination	Star trackers and electronics	Alignment criticality Interface complexity
Attitude Control	Reaction jet quad (4), storage tanks, valves and associated plumbing	Consumables, interface complexity, safety, contamination, failure rates
	Control electronics	Failure rates Technology advances
Thermal Control	Distributed Cold plates Radiators	Impractical to modularize. All passive systems selected considered as part of basic structure
Data Processing	Central processor	Failure rates, technology advances, changing requirements
Tracking and Telemetry	Central processor	Failure rates, technology advances, changing requirements

The total number of modules identified is 14. The baseline common module has provisions for 12 modules. Examination of the modules identified in the above table indicates that the data processing and tracking/telemetry modules could be combined. Also, the control electronics for attitude control and stabilization can be (and normally are) grouped together. Thus, from a servicing standpoint the support functions can be conveniently grouped into 12 modules.

In Section 3.3 (Subsystems Synthesis) the trades and identification of the specific assemblies for each function are presented. The only functional module that was a marginal fit in the standardized 24-inch by 20-inch by 24-inch package was the attitude control (RCS) module. In most cases there was ample room for expansion including integral redundancy if desired. Thermal loads of the modules are well within allowable limits.

Although no specific wire count was made of each interface, a gross analysis did indicate that all module interfaces were reasonable. Connections that are noise sensitive or require critical impedance matching are all contained within a replaceable module. Physical alignment problems are minimized in this grouping concept. All three reaction wheels are in one assembly as are the three star trackers or charge-coupled devices.

Figure 3.1-8 presents a layout of the common support module. The Astromast used for extension of the solar array is required on some of the platforms to prevent the ocultation of the sun by platform mission equipment appendages. Two radiator panels are shown. Each panel is approximately 22 square feet in area. Table 3.1-2 presents a weight summary of the common support module. This configuration was derived to accommodate the requirements of the satellites in the baseline traffic model. Except for the experimental/developmental satellites in the satellite inventory this concept of a support module could be used with all the mission equipment payloads that were defined by the inventory analysis (Section 5.1, Volume IV). The concept probably could be used with the experimental/developmental payloads also, but insufficient definition of these 1980 payloads at this time preclude such an all-encompassing statement.

Sections 4.0 and 5.0 present the derivation of the geosynchronous platforms for both the baseline and new traffic models. Definition of the mission equipment for the platforms is included. Analysis of the support requirements of this equipment indicated that for all practical purposes the most stringent support requirements for the platforms are the same as those of the satellites. Thus, the common support module synthesized above is equally applicable for the platform mission equipment. One design can be employed for either programmatic option: on-orbit serviceable satellites or multifunctional platforms. The support capabilities of the common support module are summarized in Table 3.1-3. Power, impulse, and data handling are over-sized for some of the satellites and platforms. The modular packaging concept will readily facilitate scaling of capability if it is desirable.

PLATFORM MISSION EQUIPMENT MODULE

In order to provide on-orbit servicing of mission equipment in all three modes of operation, the same requirements and design considerations that were applicable to the common support module are applicable for the structure to house the platform mission equipment. Therefore, the toroid is also the preferred concept for containing the mission equipment. In essence, the basic structure of the platforms will consist of one, two, or more toroids assembled

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Table 3.1-2. Common Support Module Characteristics

Item	Weight (lb)		Power Required (watts)	
	Subtotal	Total	Subtotal	Total
Structure		500		--
Primary	100			
Secondary	300			
Docking Mechanisms	100			
Electrical Power		1251		70
Solar Array Assemblies (2 KW)	240		10	
Power Conditioner	144		20	
Battery Pack	392		40	
Solar Array Booms	300		--	
Cabling	175			
Data Handling		70		28
Electronics	60		28	
Antenna	10		--	
Attitude Stabilization & Control		429		120
Reaction Wheels	75		30	
CCD Star Trackers	30		20	
Flight Control Electronics	44		50	
RCS Quads (4)	280		20	
Thermal Protection		39		
Radiators	26			
Cold Plates, Heat Pipes, etc.	13			
Totals		2289		218

with the common support module toroid. External viewing sensors will require some modifications to the toroid, but the equipment rings are essentially the same.

Table 3.1-3. Common Support Module Capabilities

Function	Capability
Electrical power	2 kilowatts, continuous duty
Stabilization	1 arc second/second
Pointing	10 arc seconds
Attitude control	3-axis stabilized
Impulse	39,000 pounds/second
Data handling	50 Mbps

Data Relay Platforms

The synthesis of the mission equipment for the data relay platforms is presented in Section 4.0. In no case are all 12 equipment ring openings required. Various concepts were evaluated to determine the practicality of a reduced mission equipment ring. In order to accommodate all three servicing modes, and specifically the auto-remote mode, the center volume of the common support module must remain open. Therefore, some type of structure must be added to the support equipment ring. One concept that was reasonable is illustrated in Figure 3.1-9. Some weight reduction could be realized but preliminary analysis indicated that just development of a second structural configuration would be more costly than producing the required quantity of admittedly oversized equipment rings. The additional weight to orbit at delivery did not appear to compensate for the delta costs.

Growth potential was an additional consideration. The standard equipment ring can accommodate additional functions and could incorporate total redundancy of mission equipment.

Sizing of the data relay platform mission equipment was intentionally restricted to 24-inch by 20-inch by 24-inch replaceable modules. Some of the K_{HI}-band transponder assemblies associated with the new traffic model data relay platforms exceeded the dimensional limits. This particular module required a 36-inch depth dimension. Because of the limited number of modules in the mission equipment ring it was determined that the added length of K_{HI}-band transponder was tolerable. In addition, more detailed design could result in reduction in the depth dimension. Technology advances could reduce the size. Provisions for on-orbit reassignment of all communication channels was the principal reason for the large size. With only limited fixed channel assignment the transponder in question could be subdivided into two or more standard-sized replaceable modules.

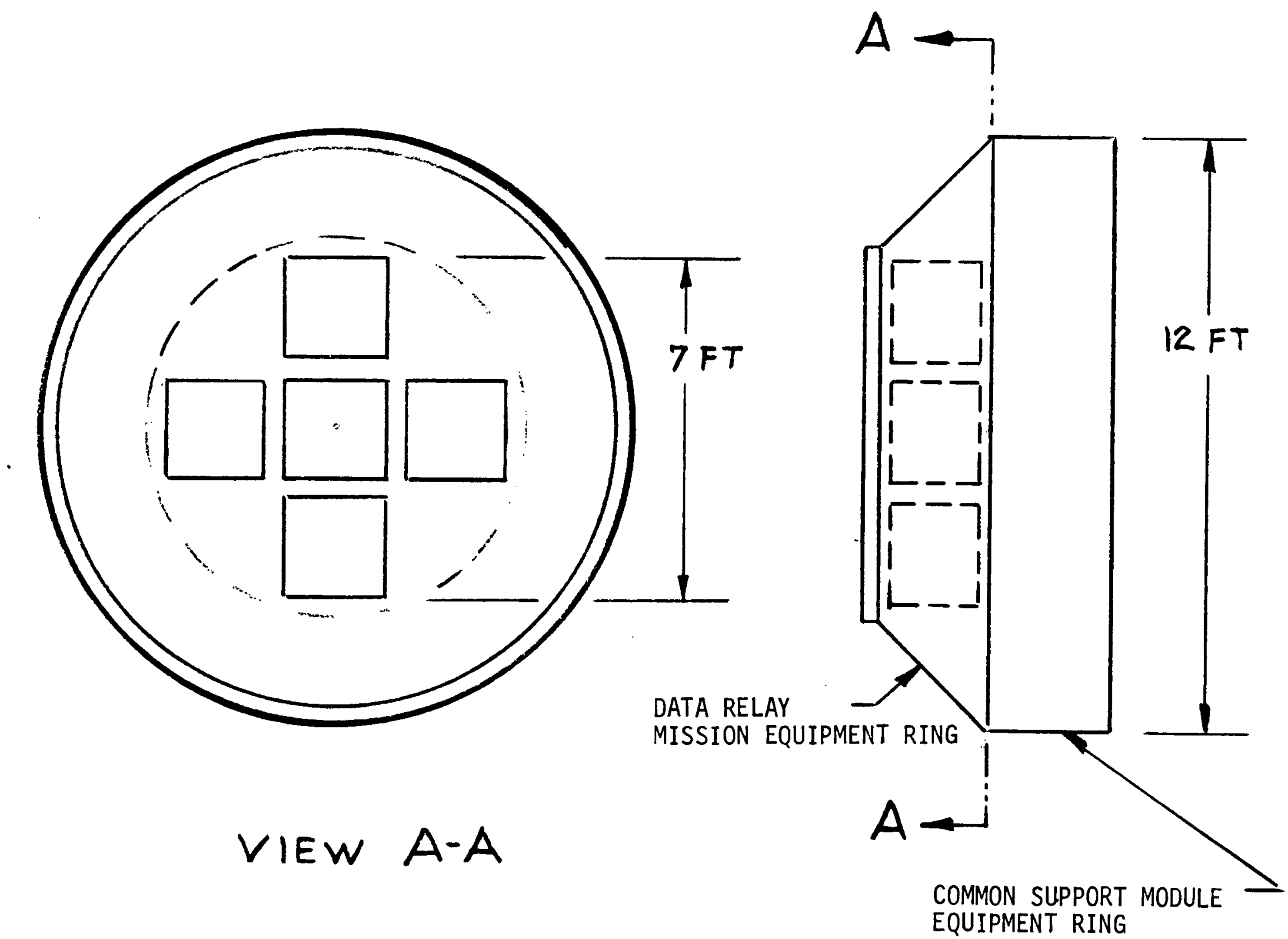

Figure 3.1-9. Modified Mission Equipment Module

The preferred configuration for all data relay platforms is two standard equipment rings, one for support systems and one for mission equipment. Module openings in both rings are also standard. There are 12 openings that accommodate a 24-inch by 20-inch by 24-inch module.

Observational Platforms

The mission equipment associated with the observational platforms present some unique requirements. The size of the modules is compatible with the standard opening. However, several of the sensors require external viewing. The accommodation approach for these sensors is to provide viewing windows or cutouts in the equipment ring for direct pressure-sealed installation of the sensor viewport.

Some of the mission equipment of the observational platforms have high power dissipation requirements (~500 watts). Special provisions must be included to assure adequate thermal contact between the replaceable module and the coldplate of the basic structure. A thermal grease packet concept that is mounted on the back of the module and is semi-crushed upon installation of the module will adequately provide the required thermal conduction.

Section 5.0 presents the synthesis of the observational platforms. No constraints to the adaptability of the standard equipment ring are identified. The anticipated complement of mission equipment utilizes most of the openings in the equipment ring. In fact, in some cases as many as three rings are required.

Some of the sensors require boom structure to extend them out from the platform. Others required mounting on a turret to permit sequential use in conjunction with a specific telescope. These provisions are also delineated in Section 5.0. In no case do these special mounting provisions negate the use of the common equipment ring. The standard toroid is used for both the subsystem support equipment and the mission equipment on the observational platforms.

3.2 SUMMARY OF SUPPORT SYSTEM REQUIREMENTS

The satellite inventory analyses presented in Volume IV indicated a potential commonality of support requirements across the spectrum of geosynchronous satellites. The previous section, 3.1, developed a common structural configuration to house both mission and support system equipment. Sections 4.0 and 5.0 of this volume develop the requirements for geosynchronous platforms. This section presents a compilation of all the support requirements for geosynchronous space elements. In essence, this section serves as an introduction to the subsystems synthesis in the next section.

SATELLITE SUPPORT REQUIREMENTS

A broad range of satellite capabilities and requirements are identified in the satellite inventory analysis. Power requirements range from a few hundred watts to as much as 2 kilowatts. Continuous duty operations as well as periodic operations impose stringent requirements on both the energy storage concept (for eclipse periods) and the thermal control concept. Pointing requirements vary from a few tenths of a degree to as low as 0.1 arc second. Stabilization requirements are as low as 0.5 arc second/second.

For purposes of commonality (minimum development costs) the most stringent requirements that could be met with a centralized support system were selected. These requirements and the satellite driver are summarized in Table 3.2-1. Sensors that require more stringent pointing and stability performance can be accommodated either by means of an independent control loop about the sensor (independent gimbaling) or by utilizing the sensors as an input source to the central control system.

Table 3.2-1. Satellite Support System Requirements

Function	Requirement	Satellite Driver
Power	2 kilowatts	Earth observation
Telemetry	50 Mbps	Astronomy, earth resources
Command	2 kbps	All
Pointing	10 arc seconds	Astronomy
Stabilization	1 arc second/sec	Astronomy
Propulsion	12,000 lb-sec	Earth observations
Thermal control	40 F to 70 F	

PLATFORM SUPPORT REQUIREMENTS

There are eight types of geosynchronous platforms that are synthesized in Sections 4.0 and 5.0 to fulfill the requirements of the baseline and new traffic models. The support requirements for each type of platform are discussed below.

Data Relay Platforms

All the data relay platforms for both traffic models and the four global regions have the same support requirements except for power. The only significant differences in the platforms are the antenna configurations and the complement of transponders. Power requirements range from 600 watts (Region I baseline traffic) to 1455 watts (Region IV baseline traffic). Table 3.2-2 summarizes the most stringent support requirements of the mission equipment of the data relay platforms.

Table 3.2-2. Data Relay Platform Support Requirements

Function	Requirement
Power	1500 watts
Telemetry	None (included in mission equipment)
Command	64 bps
Pointing	0.2 degree
Stabilization	0.1 degree/sec
Propulsion	10,000 lb-sec
Thermal control	40 F to 70 F

Tracking and Data Relay System Platform (TDRS)

The two support requirements of the TDRS that are different from data relay platforms are power and propulsion. Only 240 watts of power are required by the mission equipment. The unique requirement to include station change capability of the platform of up to 65 degrees in longitude increases the propellant requirements to over a 12,000 lb-sec impulse.

Navigation and Traffic Control Platforms (New Traffic Model Only)

All requirements of the navigation and traffic control platforms are less stringent than those associated with data relay platforms.

Earth Observation Platforms

The most significant support requirements of the earth observation mission equipment are presented in Table 3.2-3. The power requirement is based upon sequential operation of some of the sensors. No requirement could be established for all sensors to be operating at any one time. Data rates reflect the requirements of imaging cameras. Both wide field-of-view and narrow field-of-view instruments are included in the mission equipment. This established the dual pointing/stability requirements.

Table 3.2-3. Earth Observation Platform Support Requirements

Function	Requirement
Power	1600 watts
Telemetry	50 Mbps
Command	2 kbps
Pointing*	1 arc second
Stability*	1 arc second/second
Propulsion	16,500 lb-sec
Thermal control	40 F to 70 F

*Majority of sensors require this accuracy; sensors viewing through telescope require 1 arc second (per second) accuracy and must be tied into the control loop.

Solar Astronomy Platform

The most stringent requirements in two support functions are established by the solar astronomy platform. The functions are pointing/stability and propulsion. The desired control accuracies require not only sensor input to the control system but also special gimbaling of the sensors themselves. Table 3.2-4 summarizes the requirements of this platform.

Table 3.2-4. Solar Astronomy Platform Support Requirements

Function	Requirement
Power	1328 watts
Telemetry	40 Mbps
Command	2 kbps
Pointing	0.1 arc second
Stability	0.05 arc second/second
Propulsion	19,000 lb-sec
Thermal control	40 F to 70 F

Stellar/X-Ray Astronomy Platform

All support requirements of the stellar/X-ray astronomy platform are comparable to those of the solar astronomy platform with the exception of power. Only 630 watts of power are required for the mission equipment.

Plasma Physics Platform

Support requirements for the plasma physics platform are quite nominal. The requirements are summarized in Table 3.2-5.

Table 3.2-5. Plasma Physics Platform Support Requirements

Function	Requirement
Power	1280 watts
Telemetry	100 kbps
Command	1.0 kbps
Pointing	2.0 arc minutes
Stability	1.0 arc minute/sec
Propulsion	7200 lb-sec
Thermal control	40 F to 70 F

High-Energy and Magnetophysics Platform

Of all the observational platform types this platform has the least stringent requirements except for propulsion. It is about the heaviest platform in the inventory. Also, the type of mission equipment on this platform does not lend itself to grouping with other equipment because of the required magnetic fields generated by the equipment. The support requirements are summarized in Table 3.2-6.

Table 3.2-6. High-Energy and Magnetophysics Platform Support Requirements

Function	Requirement
Power	580 watts
Telemetry	50 kbps
Command	1.0 kbps
Pointing	1.0 degree
Stability	0.5 degree/second
Propulsion	18,000 lb-sec
Thermal control	40 F to 70 F

CONCLUSION

The most stringent mission equipment support requirements for the inventory of eight different types of platforms are presented in Table 3.2-7. Inclusion of the power consumption of the supporting subsystems will result in a total power load of approximately 2 kilowatts. The pointing and stability requirements reflect the necessity of a direct input from the sensors into the control loop. Centralized support system capability actually is of the order of 10 arc seconds pointing and 1 arc second/second stability because of technology limits, structural flexure, structural alignment, and thermal gradient distortion.

Table 3.2-7. Platform Support System Requirements

Function	Requirement	Platform Driver
Power	1600 watts	Earth observation
Telemetry	50 Mbps	Earth observation
Command	2.0 kbps	Earth observation
Pointing	0.1 arc second	Solar/stellar astronomy
Stability	0.05 arc second/second	Solar astronomy
Propulsion	19,000 lb-sec	Solar astronomy
Thermal control	40 F to 70 F	All

Several of the representative sensors that were defined for the observational platforms are laboratory equipment. The temperature limits of some of this equipment is more stringent that the proposed control range of 40 F to 70 F. All equipment was rated for operation at 70 F. However, several items were listed as requiring higher temperatures than 40 F. For purposes of this study it is assumed that either the space-rated equivalent of the items in question can operate down to 40 F or heaters would be included in the equipment to maintain the desired minimum temperature.

Comparison of Tables 3.2-1 and 3.2-7 indicates that the only difference in the support requirements for the satellite inventory and the platform inventory is the propulsion or impulse requirements. Not only are the weights of the observational platforms significantly larger than the individual satellites but also the mean mission durations of the platforms were assumed to be 5 to 7 years as compared to 3 to 5 years for the satellites.

The subsystem performance requirements for the common support module for both satellites and platforms are essentially the same and therefore are as listed in Table 3.2-7. Only one common support module concept is required. It will be the structural configuration developed in Section 3.1. Subsystem modules are to be packaged in 24-inch by 20-inch by 24-inch replaceable modules.

3.3 SUBSYSTEMS SYNTHESIS

Trade studies on alternate mechanizations for the various support functions required by both satellites and platforms for the new and the baseline traffic models are presented in this section. Sections 3.1 and 3.2 indicated that the subsystem requirements for satellites and platforms were in the same range and thus, commonality of equipment was feasible. Because of this commonality the selected concepts are not necessarily optimum for any one space element.

The subsystem assemblies are grouped in standardized replaceable modules (24 inches by 20 inches by 24 inches). Weight, power, and volume of each module and a gross packaging layout are also defined. Where appropriate, plateaus of subsystem capability and consequently subassembly groups are defined. The more significant selected concepts for the subsystems are as follows:

Electrical power	
Power generation	Solar arrays
Energy storage	NiCd batteries
Data handling	
Telemetry	Ku-band
Attitude stabilization and control	
Stabilization	Reaction wheels (3)
Pointing	Change coupled device star trackers (3)
Propulsion	Hydrazine; 4 quad; 16 engine
Thermal control	
Heat transport	Heat pipes

The selected concepts include provisions for at least "fail-degrade" reliability. The solar array-battery concept is inherently redundant. Total redundancy is provided in the data handling mechanization. Failure of a single reaction wheel will degrade performance and increase propellant consumption. Only two of the three star trackers are required for pointing and attitude reference determination. Redundant modules are included. The heat pipe concept is inherently reliable because of its passive nature.

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ELECTRICAL POWER SUBSYSTEM

This section of the report presents a comparison of alternate electrical power subsystems (EPS) that are applicable for geosynchronous platforms. The various types of platforms derived require electrical power ranging from 500 watts to approximately 2000 watts. A preferred concept that would facilitate incremental EPS capability is defined. Packaging of EPS equipment is also defined.

Solar arrays with NiCd batteries are the preferred concept primarily because of lower cost and lighter weight as compared to other concepts. The EPS is packaged in three replaceable modules: (1) power conditioning equipment, (2) battery pack, and (3) solar array and associated deployment/orientation mechanisms.

EPS Requirements

The EPS concept must have a rated life (expendables) of 10 years. Both manned and auto-remote maintenance is planned but is intended only to refurbish/replace random failures. At geosynchronous orbit, eclipse periods of up to 72 minutes will be encountered. Data relay platforms are required to maintain full operational capability during this period. Some of the operational platforms may utilize the eclipse period for unique observations. Therefore, the EPS should be designed for normal operation (full power load) on a continuous basis.

In order to reduce potential development costs the selected concept shall be defined at the following incremental levels: 500, 1000, 1500, and 2000 watts.

EPS Trades

Figure 3.3-1 shows alternative electrical power subsystem trades considered for geosynchronous platform applications. The EPS is treated as four major assemblies--energy source, conversion, storage, and power conditioning and distribution. For the purpose of brevity the energy source and energy conversion are discussed below.

Power Generation

A large number of power generation concepts can be synthesized by use of all possible combinations of energy sources and energy conversion devices. However, the trades made herein will primarily consider those on which the bulk of development effort has been accomplished. These are radioisotope and nuclear thermoelectric and solar photovoltaic generation.

Radioisotope Power Generation. Figure 3.3-2 shows weight versus power for many of the radioisotope electrical power generation devices developed or studied during the past 15 years (Atomics International data). As can be seen, the specific power is consistently in the range of 0.5 to 2.0 watts/pound over the entire range of power from a few watts to 10 kilowatts. The shield weights and criteria for the various power systems vary considerably from instrument

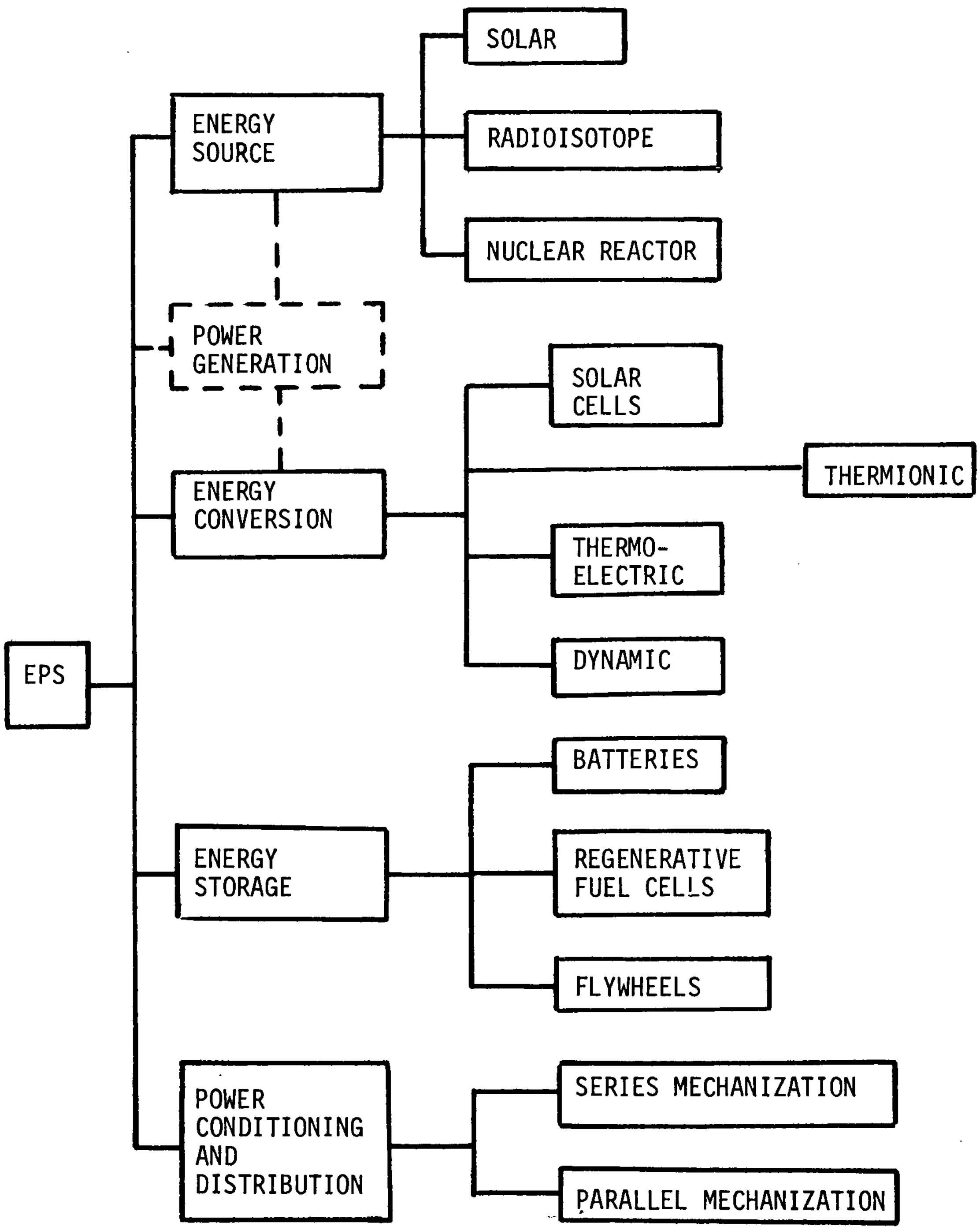


Figure 3.3-1. EPS Trade Tree

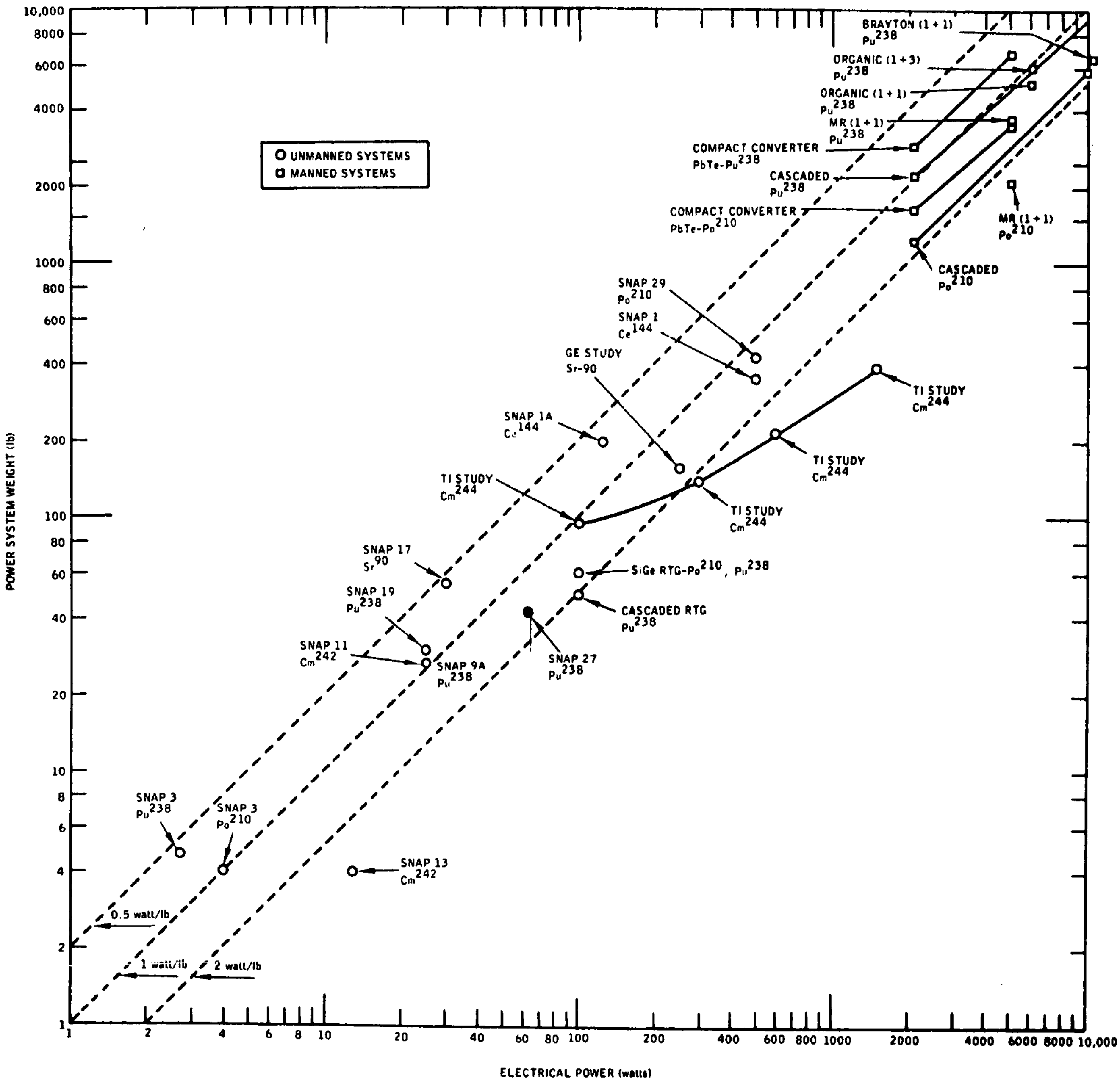


Figure 3.3-2. SNAP Isotope Power Systems

shielding in the low watt range to manned shielding in the kilowatt range. Reentry protection also varies from burnup reentry to capsule and heat source protection over this entire power range. Notwithstanding the effect of these variables, the specific weights remain in the above range with increasing power for a variety of isotope power systems.

The energy sources are based on the various radioisotopes considered applicable to space power systems. There are isotopes meeting the criteria of either minimum shield weight, long half-life, or minimum cost. No one radioisotope is optimum for all these criteria. The energy sources shown in Figure 3.3-2 are based on:

Plutonium-238 (P_u-238)

Polonium-210 (P_o-210)

Curium-242 (C_m-242)

Strontium-90 (S_r-90)

Curium-244 (C_m-244)

Cerium-144 (C_e-144)

The radioisotope systems that have been flown all use Plutonium-238.

The radioisotope power systems shown by Figure 3.3-2 incorporate several different types of energy conversion devices, including both static and dynamic methods. The static devices are either thermoelectric or thermionic. The dynamic concepts are based on the Brayton and Rankine cycles. Power systems are shown that use either organic liquids or mercury for Rankine cycle working fluids.

Radioisotope fuel capsule reentry protection varies for the various power systems shown by Figure 3.3-2. Design criteria vary from allowing the fuel capsule to burn up and disperse in the upper atmosphere to complete containment of fuel over the entire flight operational range. Dispersion of radioisotopes in the upper atmosphere is unacceptable on the basis of current safety criteria. Therefore, some of the earlier designs are unacceptable or would require a weight increase to meet safety requirements.

Table 3.3-1 (from 1971 IECEC) summarizes performance characteristics of the current family of radioisotope thermoelectric generators (RTG's). All of these power systems use P_u-238 fuel. The top portion of the table gives characteristics for the complete RTG. The lower part of the table shows weights and performance for the radioisotope heat source alone. The highest power available is 155 watts shown for the multi-hundred-watt (MHW) RTG. For the minimum geosynchronous platform power requirement of 500 watts (600 watts including losses), four of the MHW generators would be required. The MHW is currently being developed. Table 3.3-2 shows more recent data for the MHW, which indicates that power has gone down and the weight has increased. Although the design lifetime (12 years) meets geosynchronous platform requirements, it does not appear to be cost and weight competitive for geosynchronous platform application. The quoted cost of P_u-238 varies from \$500.00 to \$700.00 for a thermal watt (space station study). Assuming an average cost of \$600.00 per

Table 3.3-1. RTG Performance Characteristics

PROGRAM	SNAP 19 NIMBUS	SNAP 27 APOLLO	TRANSIT (ISOTEC)	PIONEER (SNAP 19-TAGS)	SNAP 27 INTEGRAL	MHW[5]
Converter						
Specific power (w/lb)						
Beginning of life	0.96	1.64	1.31	1.39	1.26	2.09
End of life	0.81	1.58	1.06	1.14	1.10	1.66
Power (watts)						
Beginning of life	28.1	71.0	37.0	40.9[1]	62.4	155.0
End of life	23.5	68.8	30.0	33.5[2]	545.0	123.0
Weight (pounds)	19.1	27.8	12.5	18.6	25.8	34.0
RTG total	29.2	43.3	28.2	29.4	49.5	74.0
Mission time (years)	1	1	5	2.5	5	12
Efficiency (percent)						
Beginning of life	4.5	4.8	4.62	6.4	4.46	7.05
End of life	3.76	4.65	3.75	5.23	3.83	5.6
Hot junction temperature (degrees F)	970	1070 day 1010 night	752	990	1040[3] 1000[4]	2012
Size, inches (diameter x length)	19 x 21	15.7 x 18	23 x 17	15.7 x 11.2	12.1 x 18.4	13.9 x 20
Heat Source						
Heat load, W_T (water thermal)	625	1480	800	640	1400	2200
Weight (pounds)	10.1	15.5	13.85	11.04	23.7	40.0
Specific power, W_T/lb	62.0	96.0	57.7	58.0	59.0	55.0
Operational temperature (degrees F)						
Surface	1100	1380	942	1100	NA	2150
Structure	1536	1380	1350	1880	NA	NA
STATUS	FLOWN	FLOWN	PROTOTYPE	PROTOTYPE	STUDY	PRELIMINARY DESIGN

NOTES:

1. At fueling
2. 3.25 years after fueling
3. For one-year mission
4. For five-year mission
5. SiGe T/E material

thermal watt for P_u-238 and a converter efficiency of 6.2 percent, the cost of MHW electrical power would be $9700.00 per electrical watt for the fuel alone. To this must be added the cost of fuel encapsulation, reentry protection, and the thermoelectric generator. This may be compared with solar array power costs of $400.00 to $600.00 per electrical watt.

Table 3.3-2. Multi-Hundred-Watt
Radioisotope Thermoelectric Generator

Power Level	150 w (e)
Weight	80 to 90 pounds
T/E Material	SiGe
Fuel	P_u-238
Safety	Designed to composite mission environment
Envelope	
Height	23 inches
Diameter	16 inches

For the power ranges of interest to the geosynchronous platform, a radioisotope dynamic power system would be more cost effective than the RTG's. For example, Brayton cycle thermal efficiencies are approximately four times better than the MHW (28 percent compared to 7 percent). This would reduce fuel costs by a factor of four. However, for a dynamic power conversion system it would be difficult to obtain the required 5 to 10 years life without replacement of the rotating machinery.

Nuclear Reactor Power Generation. Nuclear reactors may be used as an energy source in combination with various types of conversion systems to generate electricity. For power in the kilowatt range, the bulk of the development effort to date has been applied to reactor-thermoelectric power systems. SNAP-10A with 500 watts electrical output was flown in 1965.

Table 3.3-3 shows characteristics for reactor-thermoelectric power system designs (Atomics International data). Two types of converters are considered. Specific weight decreases significantly with increased power output. For a 2000-watt output, a specific power of 1.7 watts per pound is indicated. This is comparable with the higher performance RTG's. However, the shadow shield weights shown by Table 3.3-3 are minimal in that they are based on allowable dosages to electronic components at 50 feet distance from the reactor. Certain experiments and payloads would require large increases in shielding weight to obtain substantially lower radiation levels.

Table 3.3-3. Reactor Thermoelectric Systems Characteristics

Converter Type	Direct Radiating SiGe			Compact Converter PbTe		
Net electrical power (kwe)	2	5	10	2	5	10
Reactor outlet temperature (deg F)	1300	1300	1300	1250	1250	1250
Reactor thermal power (kw)	85	215	430	60	150	300
Radiator area (ft^2)	100	250	500	100	250	500
Weights (lb)						
Reactor	330	400	510	320	360	440
Shield (10^{11} nvt/yr, 4.5 x 10^5 rad/yr, 50 ft separation distance)	320	435	520	300	390	475
PCS	550	1200	2390	510	1120	2170
Total	1200	2035	3420	1130	1870	3085
Specific weight (lb/kwe)	600	407	342	565	374	309

The estimated cost of reactor thermoelectric power systems (Atomics International data, 1969) varies from $500.00 to $2000.00 per watt electrical. The higher costs are related to the smaller systems of interest to the geosynchronous platform. However, to meet a 5- to 10-year operational requirement, reactor test data indicate derated operation (reactor outlet temperatures substantially lower than 1300 F). This causes a further cost increase.

Solar Energy Conversion. Solar energy may be used in combination with several different types of conversion devices to generate electrical power. Only a small amount of effort has been applied to the development of heat-power devices using solar concentration in combination with Rankine or Brayton cycle conversion machinery. Static conversion using thermoelectrics and thermionics have also been investigated. However, the conversion of solar energy to electricity has proved to be most practical by use of solar cells. These devices convert solar energy directly to electricity. With the exception of a few RTG's, all of the unmanned spacecraft to date are powered with solar photovoltaic power sources.

Figure 3.3-3 shows solar array specific energy for several different designs. The bulk of larger solar arrays flown today are rigid panels. Solar cells are mounted on sheets of aluminum honeycomb which are hinged together and folded for stowage during launch. With orientation and deployment mechanisms the rigid arrays generate 4.0 to 8.0 watts of electricity per pound of weight. Current arrays use solar cells 12 to 14 mils in thickness. Advanced flexible lightweight arrays will use solar cells from 4 mils to 8 mils in thickness. The 1.5-kilowatt FRUSA (flexible rolled-up solar array) has been flight tested. Specific energies of up to 30 watts per pound are projected for advanced flexible solar arrays. Current solar array power sources have an approximate cost range of $400.00 to $600.00 per watt. Operational lifetimes, allowing for degradations of 10 years, are projected.

Power Source Selection. Solar photovoltaic power generation is selected for the geosynchronous platform application. This is based on cost, weight, and availability.

Energy Storage

During eclipse periods, electrical power must be obtained from energy storage devices. For long-duration missions requiring a large number of charge/discharge cycles, nickel-cadmium (NiCd) batteries are used. Since the subject missions are scheduled for the 1980 time period, two other energy storage devices were indicated in the trade tree (Figure 3.3-1). These are flywheels and integrated regenerative fuel cells.

One concept is an integrated power and attitude control system (IPACS). During the sunlight part of the orbit, solar array power is used by an electric motor to increase the speed of a flywheel. This wheel may be common to a control moment gyro. During an eclipse, energy is taken from the wheel to operate a generator to supply electrical power to the loads. IPACS is currently in a Phase A study phase.

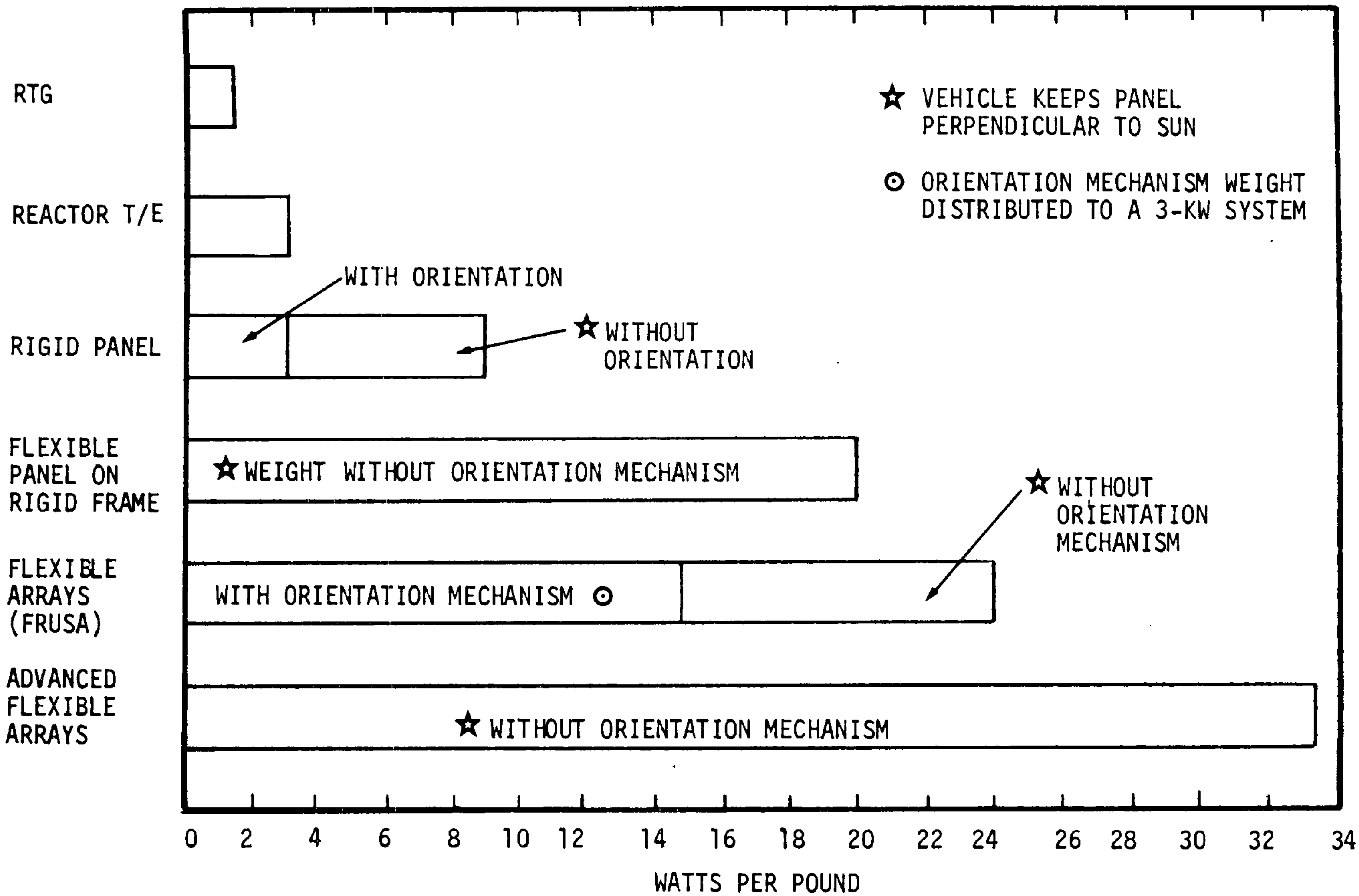


Figure 3.3-3. Prime Power Sources Figure of Merit Comparison

Figure 3.3-4 shows energy storage subsystem usable energy density as a function of orbit altitude and estimated operational lifetime.

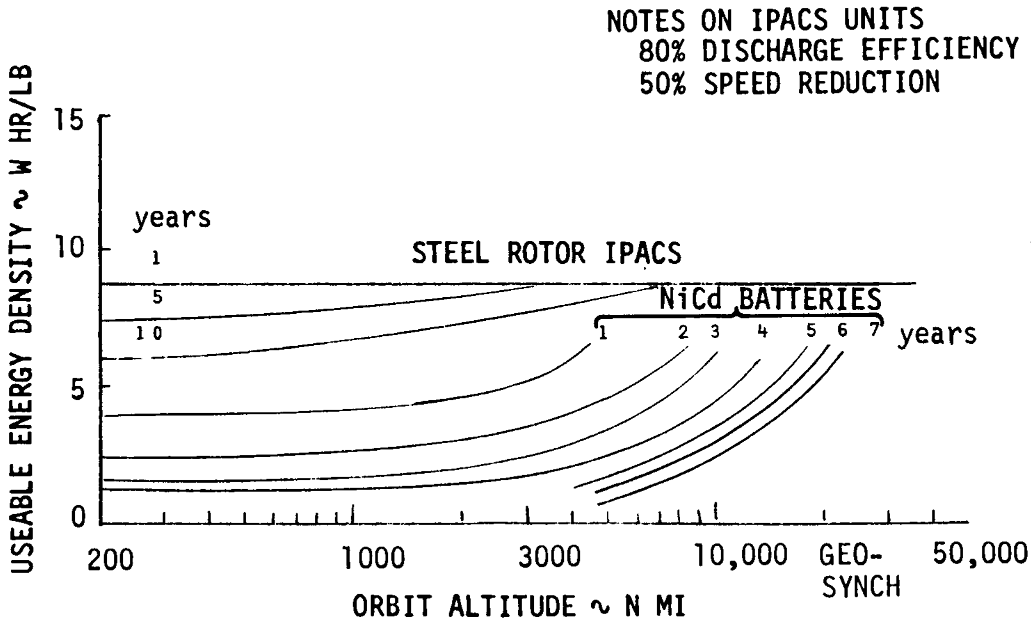


Figure 3.3-4. Energy Density Variation with Orbit Altitude and Mission Duration

The data shown for the nickel cadmium (NiCd) batteries are based on a 125-pound energy storage subsystem having a 1000 watt-hour capacity. Battery life data used for this analysis are based on tests conducted by the Quality Evaluation Laboratory of the Naval Ammunition Depot, Crane, Indiana. These data take into account long periods between cycling which occur at geosynchronous altitude. The increase in usable energy density at the higher altitudes is due to higher usable depths of discharge permitted by a reduction in charge-discharge cycles required. For example, at 200 n mi there are 5960 eclipse periods in a year and at geosynchronous there are 90.

The potential advantage of the IPACS is shown in the data for a 15 watt-hour per pound rotor operated at 50-percent speed reduction and 80-percent charge-discharge efficiency. These data represent a conservative rotor system estimate.

The IPACS advantage clearly increases with longer or lower orbit missions (increased charge-discharge cycles) as the major factor in increased cycles for the rotor occurs as a stress derating factor on the wheel. Studies showed that stress derating due to more acceleration cycles was not as significant as the battery depth of discharge weight penalties.

Figure 3.3-5 shows the energy density of regenerative fuel cells when compared to NiCd batteries. Solar array power is stored in the regenerative fuel cell by electrolysis of water contained in the cell. During the eclipse period the gaseous hydrogen and oxygen are recombined (fuel cell) to produce electrical power. This type of device is currently in the component development stage. Availability and possible lifetimes are yet to be determined.

The future use of IPACS and regenerative fuel cells for spacecraft is not clearly defined. Therefore, NiCd batteries have been selected for energy storage.

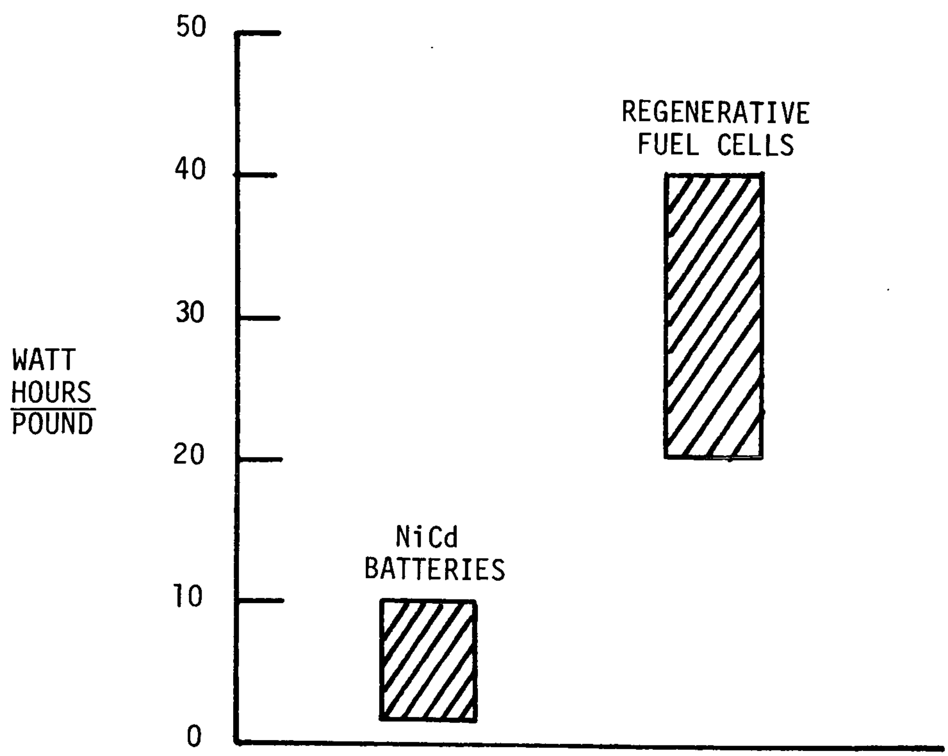


Figure 3.3-5. Energy Density Comparison

Power Conditioning and Distribution

The voltage generated by a solar array may vary by a factor of two due to the temperature excursions caused by eclipse periods. In order to simplify the design of loads and maintain a reasonably consistent performance, various amounts of bus voltage regulation are required. There are three basic approaches to solar array voltage regulation. These are shown by Figure 3.3-6.

The Type A method uses regulation in series with the solar array and the payload. This method is employed on many unmanned satellites and the manned Skylab. Its advantage is that it is rugged and simple and lends itself to redundancy (many channels). Its main disadvantage is the increase in solar array area required to compensate for the 10 to 15 percent in power losses incurred by channeling all of the load power through the regulator.

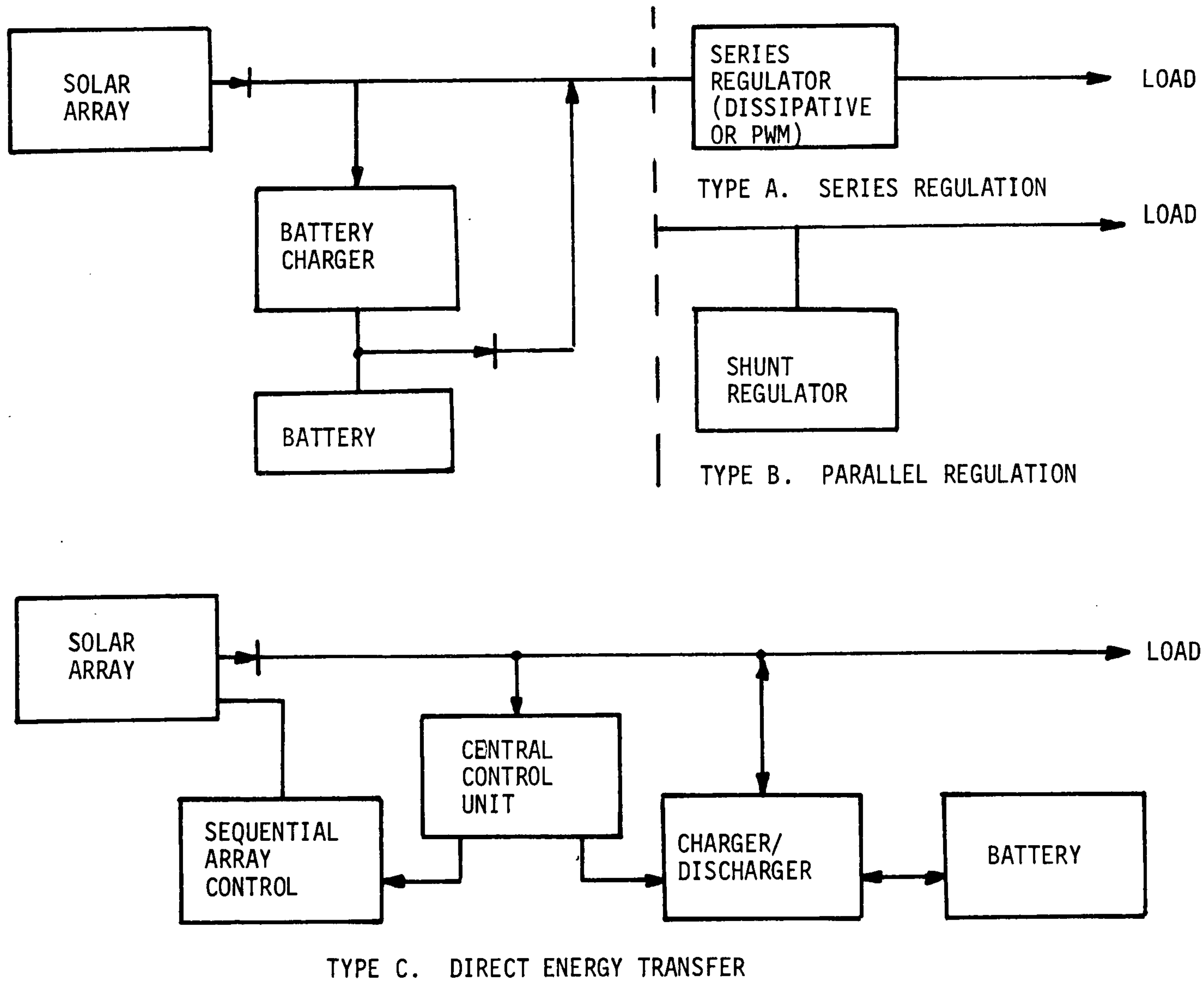


Figure 3.3-6. Power Conditioning Alternate Configurations

The Type B approach uses a shunt regulator in parallel with the solar array and the load. These also can be of the pulse width modulated or dissipative type.

The Type C method or direct energy transfer (DET) concept of power conditioning and distribution is designed to continuously supply regulated power, provide control of battery charging, and dispose of any excess power generated by the solar array. It is a refinement of the Type B approach in that a central control modulates the operation of a shunt regulator, battery charge regulators, and a battery regulator. This concept provides an efficient means for array power utilization in that no in-line conditioning element from the solar array is used. Bus voltage regulation of ± 1.0 percent (28 volts bus) is easily obtainable.

The DET system is currently being considered for the new generation of communication satellites. It is the best design from a weight and efficiency standpoint. In addition, since components for the DET system will be developed for other programs prior to 1980, it is selected for the geosynchronous platform EPS.

Selected EPS

Summarizing the results of the trade study, the selected EPS for the geosynchronous platform is based on:

- Solar photovoltaic power generation
- NiCd batteries for energy storage
- DET-type power conditioning and distribution

EPS Modularized Design for Geosynchronous Platform Use

As a first approximation, components are sized for the maximum geosynchronous platform life of 10 years. Further analysis must be accomplished to determine if this is a realistic goal.

EPS Sizing Requirements

Data developed for a communications satellite study were used to generate the power model used for this analysis. Figure 3.3-7 illustrates a generalized solar array power allocation. These values are based on beginning of life solar array power. For the case where a payload used 500 watts, power was allocated as follows:

	Watts	Percent
Payload	500	56.40
Subsystems support	35	3.95
Power conditioning and line losses	55	6.20
Battery charging	75	8.45
Solar array degradation	220	25.00
Total (BOL)	885	100.00

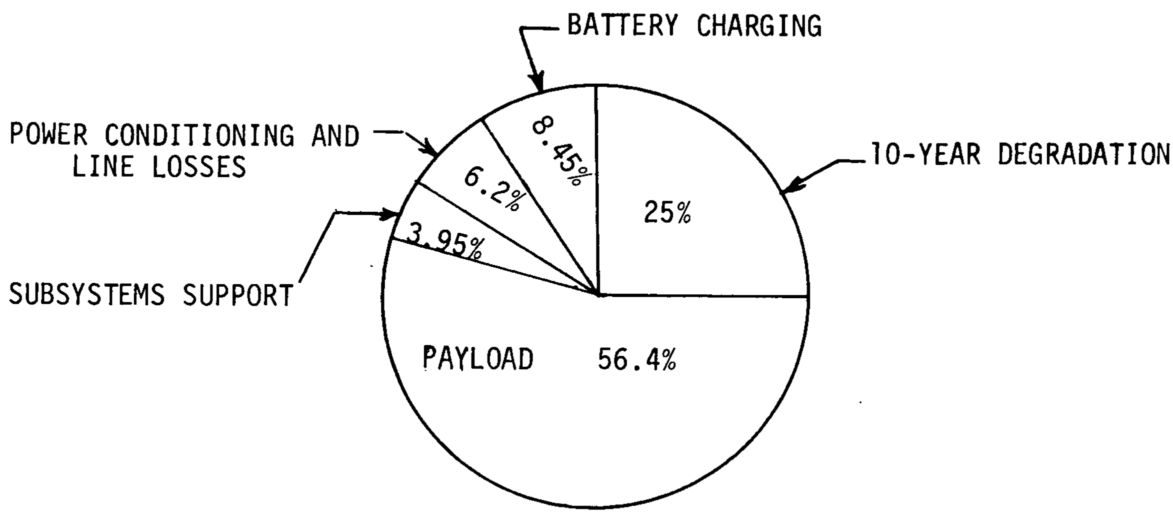


Figure 3.3-7. Geosynchronous Platform Power Allocation Based on BOL Solar Array Power

Subsystems support power includes attitude stabilization and control, reaction control system, heaters, command and telemetry, and solar panel drive and EPS central controls.

Power conditioning and line losses include diode, slip ring, and line voltage drops. Battery circuit losses are included in battery charging.

A solar array degradation allowance of 25 percent includes effects of space radiation, thermal cycling, etc.

The battery allowance is based on sequential charging of two 30 AH (17-cell) batteries at a rate of 3 amperes. A maximum eclipse period (1.2 hours) and 60-percent depth of discharge (720 watt-hours) require 75 watts of power charge at a 3-ampere rate. Allowing for battery circuit losses and battery charge efficiency, 7.4 hours are required to charge one battery.

The same sizing model is applied to payloads requiring 1000, 1500, and 2000 watts, resulting in 1770, 2655, and 3540 watts of solar array power at beginning of life.

EPS Descriptions

The EPS block diagram is shown in Figure 3.3-8. Power is supplied directly from the solar array to the loads with a central regulated 28 ±1 volt bus. Voltage regulation is accomplished by a shunt regulator operating as a variable load across lower sections of the solar array panels. By shunting only a portion of the solar array and locating the shunts on the array the net thermal

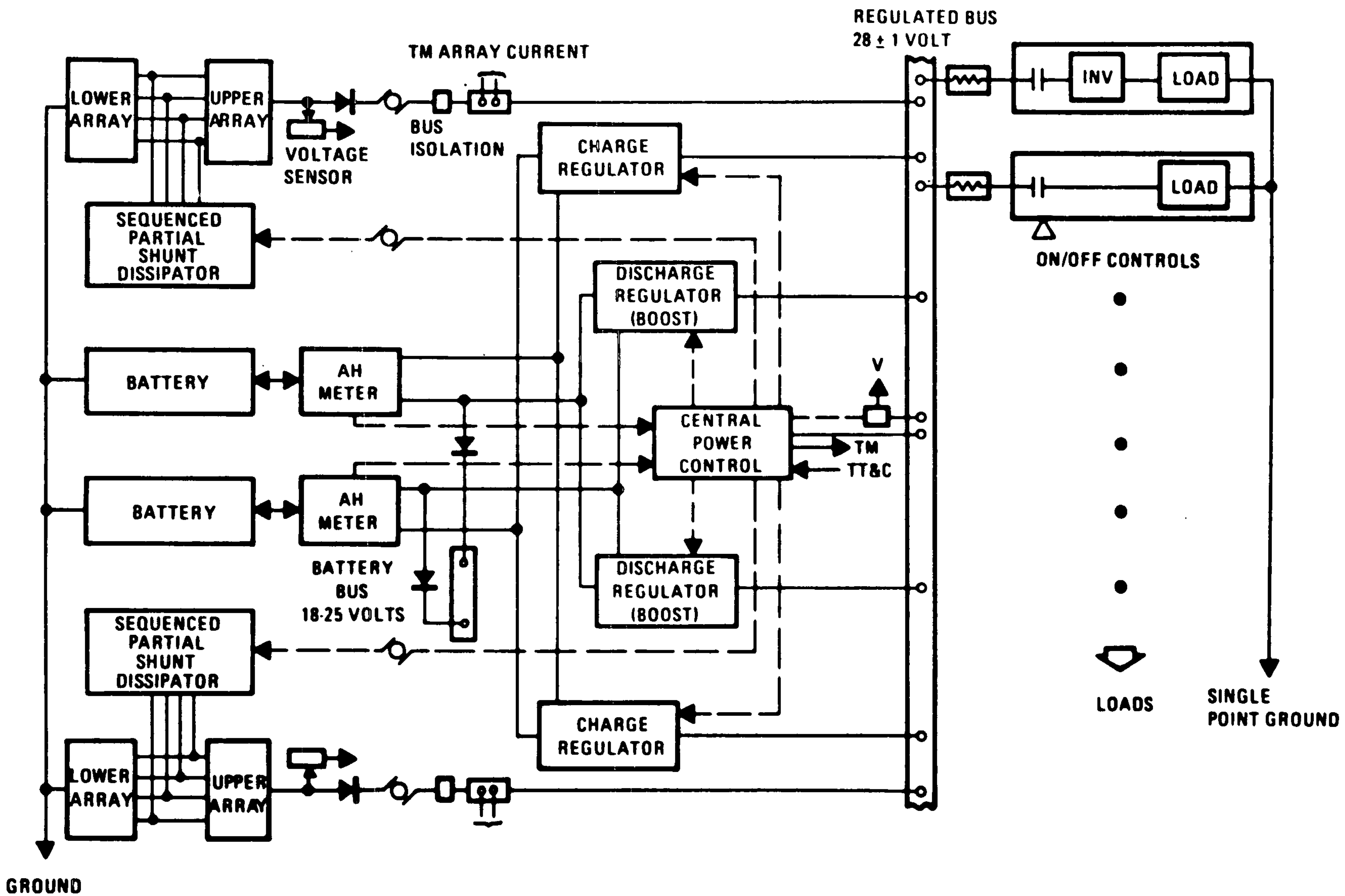


Figure 3.3-8. Electrical Power Subsystem Block Diagram

dissipation is substantially reduced. The solar array and power transfer assembly orients the solar array panels normal to, and steps it to follow the spacecraft sunline and transfer power to the central bus. Electrical power and signals are transmitted between the rotating array and the spacecraft through slip rings with redundant brushes.

The central power control unit controls the various EPS operational modes. It operates by detecting the difference between the main bus and reference voltage levels. The difference error is amplified and used to drive the system functions (i.e., supply regulated power, provide control of battery charging and to dispose of excess power generated by the solar array). Since the solar array is a constant power source, to supply larger amounts of power to subsystem loads, battery charging must be inhibited and/or the boost regulator activated to supply power from the batteries. The regulator has a nominal rating of 190 watts based on the tabulated night time load. The charge regulator is basically a series dissipative regulator controlled by external stimuli to provide desired output voltage and current conditions for charging the battery. The ampere-hour meter provides state-of-charge information for telemetry. The ampere-hour meters monitor the input and output ampere-hours of the nickel cadmium battery. This device multiplies replaced or charging power by a charge efficiency characteristics of the battery which varies with battery temperature. Each meter integrates the battery current with time and keeps track of the ampere-hours in the battery. Voltage conversion or dc-to-ac inversion will be done at the loads.

Solar Array. Data from the Flexible Rolled-Up Solar Array (FRUSA) program are used to define solar array weights for this study. This program has demonstrated the deployment, extension, and retraction of a 1.5-kilowatt flexible array assembly in an orbital environment. The solar array assembly (Figure 3.3-9) consists of a storage drum mechanism with two flexible solar cell arrays. The flexible arrays were wound on the storage drum during launch and orbit insertion. Deployment was accomplished by means of boom assemblies mounted on the storage drum structure. Key design parameters are shown by Table 3.3-4. These data were used to define the modular solar array characteristics shown by Table 3.3-5. The FRUSA BOL power is 8.15 watts/square foot (130 F)--based on the 1.5-kilowatt rating. A degradation allowance of 25 percent is assumed for 10 years of operation. Incorporating the other losses shown by Figure 3.3-7:

$$\text{Solar Array Area} = \frac{\text{Power to Payload}}{.564 \times 8.5}$$

The array blanket weights shown by Table 3.3-5 are based on 0.2 pounds per square foot. The storage and deployment weight (drums, etc.) are based on 35 pounds per drum.

For the purpose of this study a single drum and drive mechanism design is a common element for all platforms (Figure 3.3-10). A weight of 35 pounds (based on FRUSA) is assumed for each of these items. They are oversized for the smaller array area to meet loads imposed by the larger arrays.

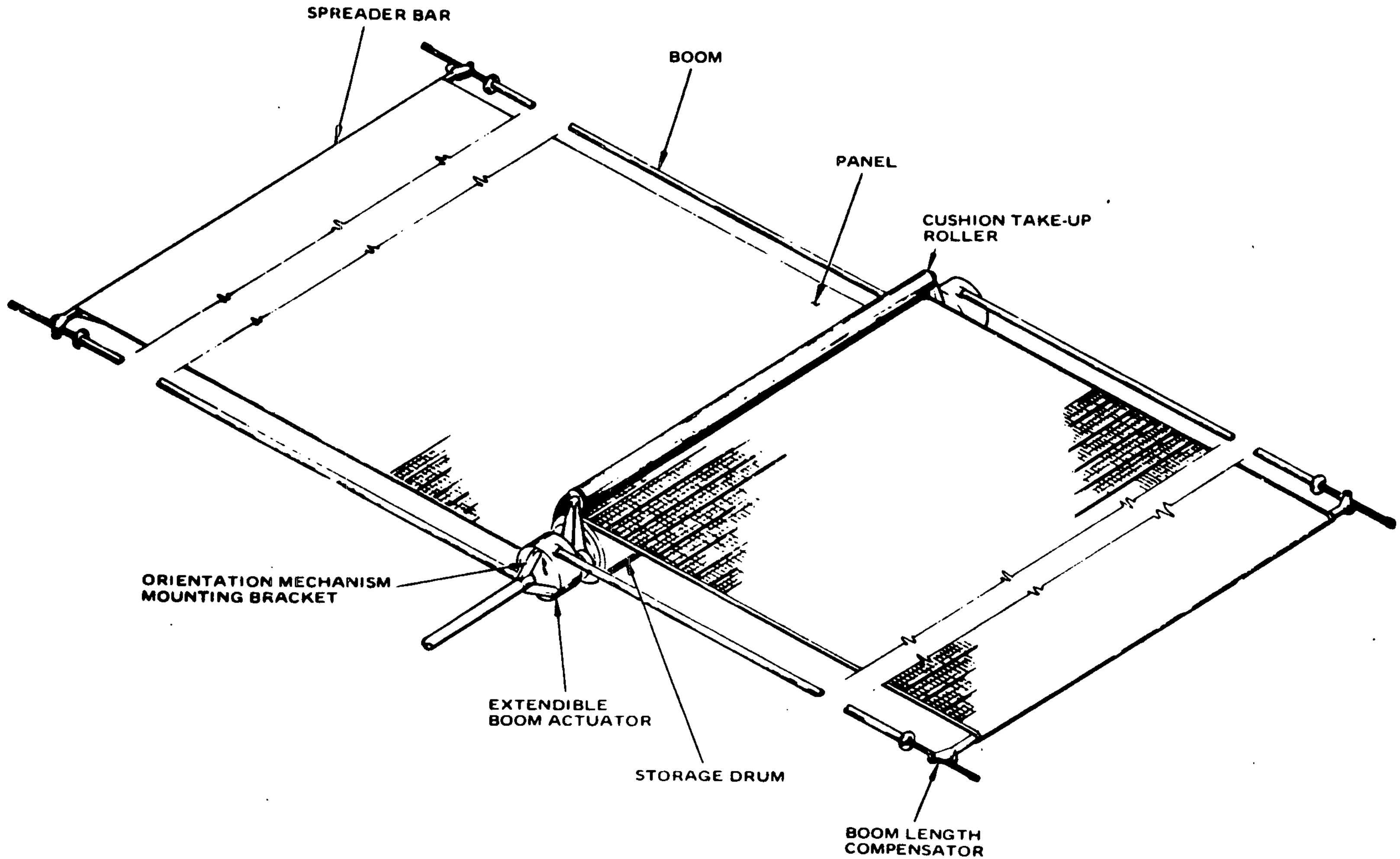


Figure 3.3-9. FRUSA Solar Array

Table 3.3-4. FRUSA Solar Array

Item	Flight Unit Status
Specific weight	46 pounds per kilowatt
Volume	2.6 cubic feet per kilowatt for 1.5 kilowatts
Power	1.54 kilowatts with full complement of cells at BOL (130 F)
Cells	7.2 mils with 6-mil coverslides
Array segment weight	0.192 pound per square foot
Orbital environment	Designed for 0.1-g quasi-static loading
Life	Operational after 8 months
Array area utilization factor	93.5 percent
Extension-retraction cycles	10 in-flight cycles completed
Solar Array Panel Area	176 square feet
Weights	Pounds
Solar array panels	34.8
Storable drum mechanisms	32.7
Array cushion, reel and drive	3.2
Total	70.7

Space Division
North American Rockwell

Table 3.3-5. EPS Characteristics Summary

Component	Weight (lb)	Dimensions (each) Length (ft)	Width (ft)	Height (ft)	Space Required Area (sq ft)	Volume (cu ft)
500-WATT SYSTEM						
Power Generation Assembly	95.0					
Solar Arrays (1)	21.0	18.9	5.5		104	
Drive Mechanisms (1)	35.0	1.0	1.0	1.0		
Storage and Deployment (1)	35.0	8.4	1.0	1.0		
Shunt Dissipators	4.0					
Miscellaneous Assemblies	234.0					
Power Conditioning and Control	36.0	1.1	1.1	0.5		
Batteries (2)	98.0	0.8	0.7	0.7		1940
Cabling Connectors	100.0					
Total	329.0					
1000-WATT SYSTEM						
Power Generation Assembly	190.0					
Solar Arrays (2)	42.0	18.9	5.5		208	
Drive Mechanisms (2)	70.0	1.0	1.0	1.0		
Storage and Deployment (2)	70.0	8.4	1.0	1.0		
Shunt Dissipators	8.0					
Miscellaneous Assemblies	398.0					
Power Conditioning and Control	72.0	1.1	1.1	1.0		
Batteries (4)	196.0	0.8	0.7	0.7		2880
Cabling Connectors	130.0					
Total	588.0					
1500-WATT SYSTEM						
Power Generation Assembly	215.0					
Solar Arrays (2)	63.0	28.4	5.5		312	
Drive Mechanisms (2)	70.0	1.0	1.0	1.0		
Storage and Deployment (2)	70.0	8.4	1.0	1.0		
Shunt Dissipators	12.0					
Miscellaneous Assemblies	561.0					
Power Conditioning and Control	112.0	1.1	1.1	1.7		
Batteries (6)	29.4	0.8	0.7	0.7		4320
Cabling Connectors	155.0					
Total	776.0					
2000-WATT SYSTEM						
Power Generation Assembly	240.0					
Solar Arrays (2)	84.0	37.8	5.5		416	
Drive Mechanisms (2)	70.0	1.0	1.0	1.0		
Storage and Deployment (2)	70.0	8.4	1.0	1.0		
Shunt Dissipators	16.0					
Miscellaneous Assemblies	711.0					
Power Conditioning and Control	144.0	1.1	1.1	2.0		
Batteries (8)	392.0	0.8	0.7	0.7		5760
Cabling Connectors	175.0					
Total	951.0					

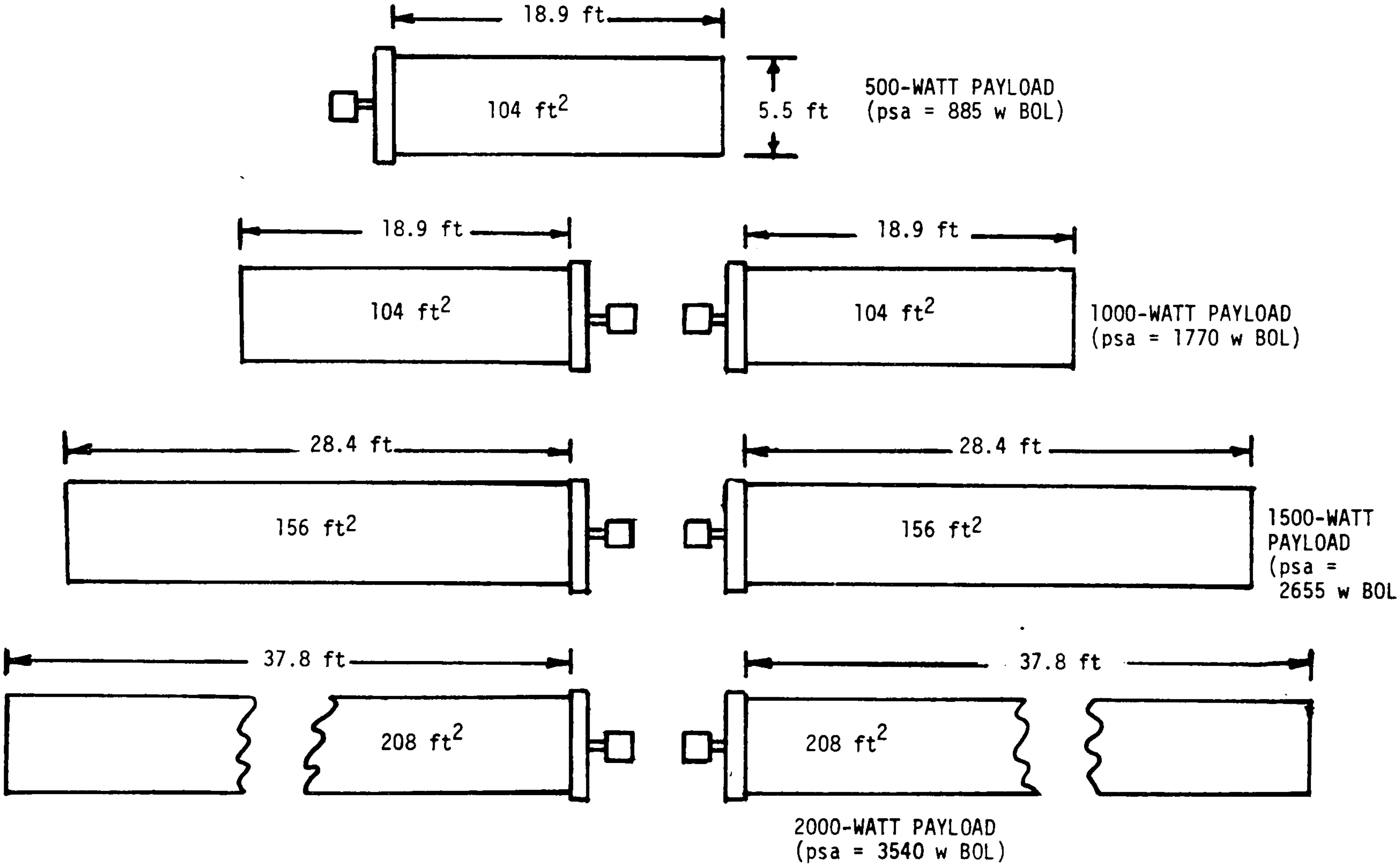


Figure 3.3-10. Deployed Modular Solar Arrays

Power Conditioning and Control. The power conditioning and control weights include two charge regulators, two discharge regulators, central power control and logic, shunt dissipation, two AH meters, voltage sensor and packaging. A weight of 36 pounds is taken from the TDRS study for a 500-watt (load) system. Cabling and connector weights are trend data values. The power conditioning and control elements can be packaged into a 14-inch by 14-inch by 6-inch enclosure for the 500-watt EPS. Multiple modules are assumed for the higher power systems.

Energy Storage. At geosynchronous orbit there are two eclipse seasons of 45 days each. Durations of eclipses vary from a few minutes at the start of period to a maximum of 1.2 hours. In order to supply 600 watts (500 watts to the payload) during the maximum ecliptic period, 720 watt-hours of battery energy are required; 900 charge-discharge cycles will be required for a 10-year lifetime. Trend data indicate that 1000 cycles can be obtained with NiCd batteries in near-earth orbit at a 60-percent depth of discharge. However, there are no test data available that show this performance can be obtained at geosynchronous conditions where cycling is spread over a 10-year period. The average depth of discharge over the ecliptic period is 45 percent if a maximum of 60 percent is assumed. To meet the requirement of 500 watts to the payload during maximum eclipse times, a battery capacity of 1200 watt-hours is required. This can be met with two NiCd batteries made up of 17 to 30 AH cells each. The characteristics of such a cell are shown in Figure 3.3-11. Allowing 25 percent for packaging, each battery will weight 48 pounds. Dimensions are 10 inches in length, 8 inches wide, and 9 inches high (volume equals 720 cubic inches). The standard geosynchronous platform container volume is 11,620 cubic inches. This space will easily accommodate the eight batteries required for the 2000-watt payload.

The weights shown in Table 3.3-5 do not include the required structure to deploy the solar array storage cylinders at various distances from the platform.

Solar Array Deployment

Depending upon the experiments carried, there will be a requirement to deploy the solar array storage cylinders at different distances from the platform. These distances vary from a minimum of 80 inches to a maximum distance of 500 inches.

After evaluation of extensible structure technology, an Astromast articulated lattice was selected for the geosynchronous platform (Figure 3.3-12). This beam lends itself to compact stowage and can be made as efficient as the basic triangular truss, with high stiffness-to-weight ratio.

In the launch mode, the entire beam is stowed within the 24-inch by 20-inch by 24-inch standard container. The extender mechanism extends the beam to the fully extended length or any fraction thereof, while sustaining the loads imposed by the solar array assembly attached to the beam tip. The deployer mechanism also retracts the beam fully or to any desired fraction of fully extended length.

CAPACITY @ 2 HOUR RATE	30 AH
CELL WEIGHT	2.3 LBS
CASE MATERIAL	304 STAINLESS STEEL
RECOMMENDED CHARGE RATE @ 25°C, 16 HOURS	3 AMPS
RECOMMENDED MAXIMUM CHARGE - CONTINUOUS @ 25°C	3.75 AMPS
TEMPERATURE RANGE - STORAGE **	-40°C to 50°C
TEMPERATURE RANGE - CHARGE	-10°C to 40°C
TEMPERATURE RANGE - DISCHARGE	-10°C to 40°C
MAXIMUM CONTINUOUS DISCHARGE	300 AMPS
IMPEDANCE	2-4 MILLIOHMS
CELL LEAKAGE RATE	LESS THAN 1X10-8 CC/SECOND OF HELIUM

RECOMMENDED -10°C to 15°C**

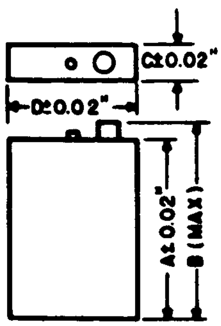


FIG-1

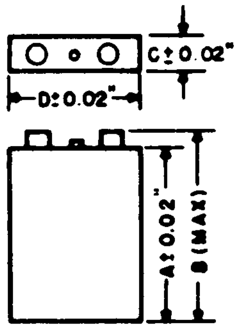
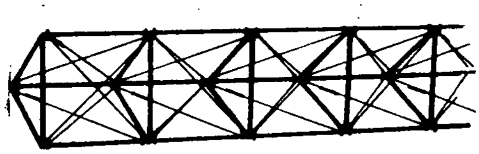


FIG-2

A	B	C	D
6.85	7.45	.903	3.00

Figure 3.3-11. 1.2 Volts Rechargeable Nickel-Cadmium Cell



TRIANGULAR SECTIONS ARE RIGID; THE LONGITUDINAL LINKS PIVOT AT EACH BAY. FOLDING IS ACHIEVED BY LOOSENING ONE TENSION MEMBER (WIRE ROPE) IN EACH BAY; THE TENSION MEMBERS ARE LOCKED AS EACH BAY IS EXTENDED.
IT IS ALSO RETRACTABLE.

Figure 3.3-12. Astromast Articulated Lattice

TELEMETRY, TRACKING, AND COMMAND SUBSYSTEM (TT&C)

The TT&C subsystem must process science and housekeeping data on geosynchronous missions to return the combined data to a controlling earth station. Also, the TT&C must receive RF command signals sent by the earth station, detect and decode the command messages, and then distribute received commands to platform subsystems and mission equipment modules.

Sizing the TT&C subsystem for the above functions is based upon the most stringent performance requirements of the entire inventory of platforms. Table 3.3-6 lists the pacing requirements. The following assumptions were made for the TT&C antenna requirements: (1) steerable antenna for mission equipment command and telemetry data links, and (2) omni antenna for support system telemetry and command data. The first assumption trades cost of the steering feature against advantages of minimal interference from and with other links or stations, minimal platform transmitter power, and minimal ground station antenna size and receiver sensitivity. The second assumption is based upon the requirement for data transfer to/from the platform regardless of orientation.

Table 3.3-6. RF Link Requirements

Mission Function	Data Rate (bps)	Bit Error Probability (Bit Error Rate)
Telemetry		
Earth resource data	50×10^6	1×10^{-5}
Ecological data	50×10^6	1×10^{-5}
Housekeeping (Engineering)	1×10^3	1×10^{-5}
Command		
Mission equipment	1×10^3	1×10^{-5}
Housekeeping	64	1×10^{-5}

Block Diagram of TT&C Subsystem

The configuration of the TT&C subsystem is shown in Figure 3.3-13. The subsystem is divided into four sections: data processing, telemetry, command, and radio frequency. The input interface to the TT&C consists of electric power, science instrument outputs, engineering sensor and transducer outputs, and the command reception omni antenna of the geosynchronous platform.

The output of the TT&C consists of an interface with all subsystems for command signals, the narrow-beam high-data rate antenna, and the low-data rate housekeeping telemetry omni-directional antenna.

Data Processing Concept

The functional requirements for the data processing section of the TT&C are assumed as follows.

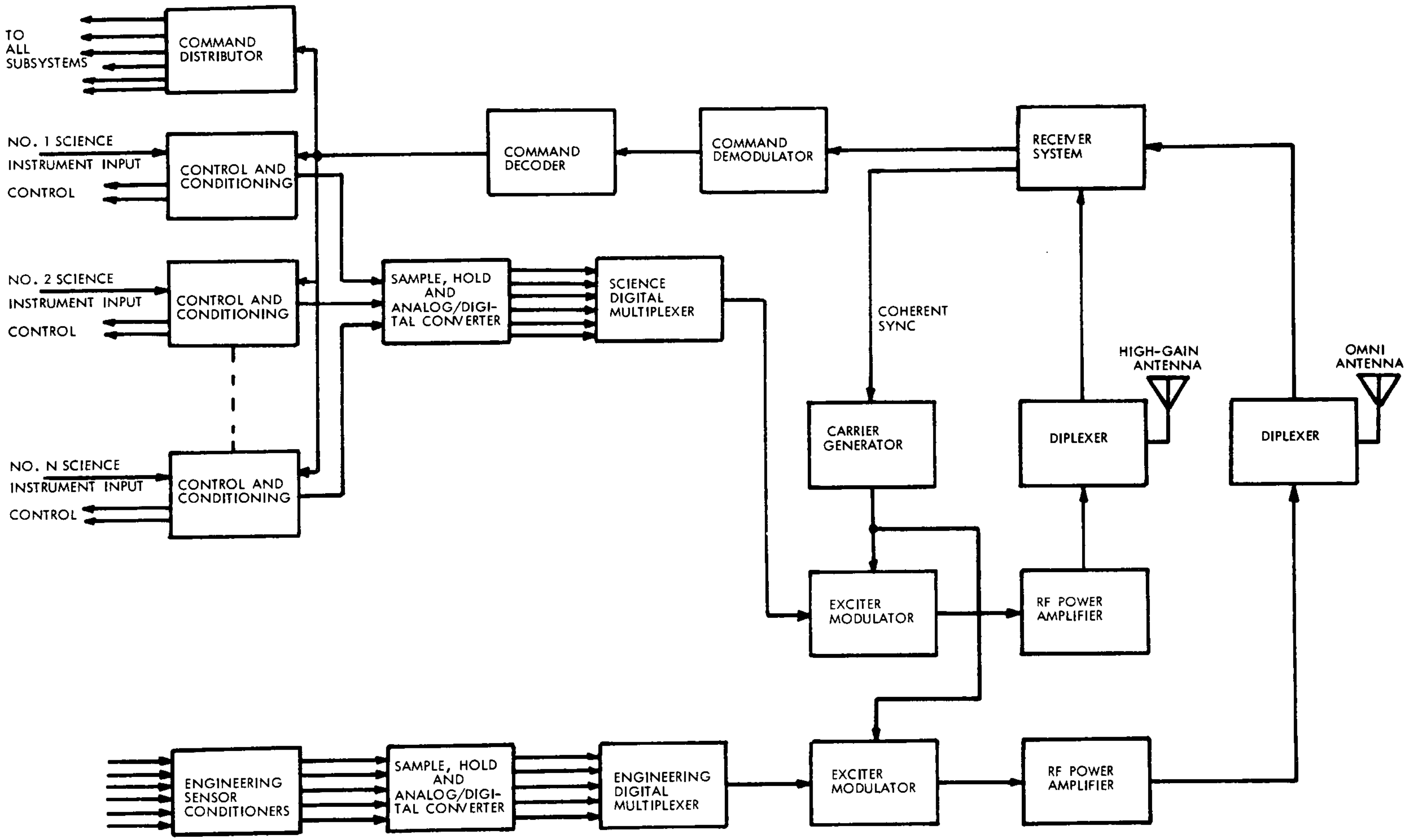


FIGURE 3.3-13. TT&C FUNCTIONAL BLOCK DIAGRAM

1. Collect housekeeping data from the support subsystems.
2. Collect science data from the science instruments.
3. Perform analog-to-digital conversion and other processing where required.
4. Format the engineering data and send it to the modulation section of the TT&C.
5. Format the science data and send it to the modulation section of the TT&C.
6. Control and sequence the sampling of the science instrument outputs.
7. Accept commands for reformatting or change in sequencing of data readouts.
8. Provide an output data rate of 50 x 10^6 bits per second.

Referring to Figure 3.3-13, the data processing section of the TT&C includes two signal conditioning/analog-to-digital converter functions, one each for science and engineering data inputs. After digital conversion, the parallel digital data outputs of the converter modules are fed to a multiplexer for time sequencing into a serial data train. Both the science and engineering data trains are fed to exciter-modulator, RF power amplifier/transmitters. The science data are transmitted through the high-gain narrow-beam antenna. Engineering data are transmitted through an omni antenna system.

Up link commands can be received by either the omni or high-gain antenna. Signal margins will limit the bit rate through the omni antenna to low repetition rates. The up link commands are demodulated and decoded and processed either to mission equipment or support system controllers.

Modulation/Demodulation Schemes

Current technology indicates that a coherent phase-shift-keyed digital modulation technique is preferred. It is anticipated that appreciable reduction in the weight and the power requirement of this concept will occur within the next decade.

Digital Versus Analog Command Techniques

The requirements considered pertinent to the command subsystem include the function to provide rapid accurate response to channel switching commands associated with a time division multiple access system (TDMA). The ultimate use of programmed channel/beam switching techniques on the platform implies need for command data in serial trains to be transmitted to the platform on a daily, weekly, and monthly basis. These considerations dictate that the scheme for command signaling must afford maximum achievable reliability with the capability to send large volumes of data in a relatively short time. The possibility for rapid command signaling on an hour-to-hour basis exists.

The use of tone command signals during the initial phase of the space program was convenient for the sending of an occasional signal to a satellite for time-independent mission commands. The need for an expeditious transmission of a lengthy train of serial data extends evaluations for candidate systems to techniques that afford rapid signal acquisition by the on-board command receiver, with increased reliability sufficient to eliminate return signal verification before command execution. A comparable technique of this type can be represented by that used on deep space surveillance satellites such as the Mariner series. Contemporary Mariner command techniques use a PN code transmitted on a separate subcarrier for bit synchronization with a one bit per second PCM/PSK/PM command channel. The time required to synchronize the command data stream is approximately 600 bit times or 10 minutes (Mariner-Mars, 1971).

Newer single-channel analog schemes eliminate the separate synchronization channel to permit more power allocation for the data channel. At a command rate of 4 bits per second, these schemes permit bit sync acquisition times as low as 35 seconds.

A further improvement must be realized to afford operation of sophisticated TDMA communication systems on a timely basis.

An all-digital, single-channel command system provides further improvement over the analog single channel scheme. The primary advantages are:

1. Higher reliability due to use of digital in lieu of analog circuitry.
2. For a 1×10^{-5} bit error probability, the signal-to-noise ratio improves 1.0 dB over analog single-channel techniques.
3. The digital scheme affords flexibility to permit use of several data bit rates in one system.
4. The size, weight, and power can be minimized by use of medium scale integration and CMOS logic circuitry.
5. The bit sync acquisition time improves to less than 11.5 seconds.

In summary, the single-channel, all-digital (detection and decoding) command scheme affords visualization of precision time control of communication channel switching and programming by the earth station.

Science Command and Telemetry Concept

The telemetry system for the science equipment must be sized to provide transmission at a data rate of 50 megabits per second at an accuracy of 1×10^{-5} bit error probability. It is assumed that (1) the earth station is limited to use of a 30-foot-diameter parabolic reflector antenna; (2) two-way coherent tracking of the platform transmission can be accomplished at Ku-band (15.3 GHz) at a system noise temperature of 60 K, equivalent G/T = 43.5 dB.

Ku-band was chosen for the high data rate (50 Mbps) for science telemetry for several reasons. S-band ground stations are not presently capable of processing over 1.0 Mbps data rates and would require significant modifications to expand their capability. Also, current frequency band allocations limit the available bandwidth to 30 MHz (2270 to 2300 MHz). Even with expanded bandwidths, overcrowding of S-band is anticipated just from the projected NASA and DOD low earth orbit payloads.

The margin calculations for a conventional Ku-band PCM/PSK/PM system are shown in Table 3.3-7. The requirements show that an RF power of 500 milliwatts is necessary. A solid-state transmitter with an overall dc-to-RF conversion efficiency of 20 percent is visualized for operation at the 50-megabit rate. This places the input power requirements for the transmitter at approximately 2.5 watts. The 14.4 dB margin affords sustained transmission during heavy rainfall.

The science command link has the potential for a significantly larger signal margin. Only a 1.0 kbps data rate must be transferred and a ground transmitter at Ku-band does not have the power limitations of a platform transmitter. No problems are anticipated with the science command link.

Engineering Command and Telemetry Concept

Both the command and telemetry links for engineering or housekeeping functions must be maintained during both normal and failure mode conditions. The ground/platform data interchange must be independent of platform orientation. Redundant equipment is mandatory. This design criterion is imperative in order to support the on-orbit servicing mode of operation. In the event of a platform failure, that failure must be capable of being diagnosed at least to the replaceable module level in order to deliver the appropriate replacement equipment. In certain failure modes it will be necessary to initiate platform actions (maneuvers, power down, self-test, etc.) in order to identify the faulty equipment and possibly activate an alternate or backup mode of operation.

S-band has been used almost exclusively for the relay of housekeeping data on past space programs. However, Ku-band is the preferred mode of operation for platforms. The science data relayed via Ku-band, and thus a simplification in the ground station can be achieved by also relaying housekeeping data via Ku-band. The bandwidth requirements are well within the capability of the ground equipment. A single antenna can be used for both reception and transmission of both classes of data. Similarly, a single wide-band receiver front end can be used. The carrier frequencies of the links can be established within a 200 MHz band and provide adequate bandwidth for the encoded data. Antennas and receivers with 200 MHz bandwidth at Ku-band are achievable.

A comparison of S- and Ku-band systems for the housekeeping function was conducted. Margin calculations for the command link at S- and Ku-bands are summarized in Tables 3.3-8 and 3.3-9, respectively. The low data rate required for housekeeping commands result in minimal power requirements for the ground transmitter at either frequency. Only a 64 bps command data rate was identified, but a much larger data rate could be accommodated by increasing the transmitter power to several watts.

Table 3.3-7. TT&C Ku-Band 50 Mbps Downlink Margin Calculation - Geosynchronous Altitude

Parameter	dB*
1. Total transmitter power, 500 mw (dBm)	+27.0
2. Transmitter circuit loss	-0.5
3. Transmitting antenna gain, 3.0 ft dia	+41.1
4. Transmitter antenna pointing loss	-1.0
5. Space loss, F = 15.3 GHz; D = 19,300 n mi	-206.3
6. Polarization loss	-0.1
7. Receiving antenna gain, 30 ft dia	+61.3
8. Receiving antenna pointing loss	0.0
9. Receiving circuit loss	0.0
10. Net loss (2 + 3 + 4 + 5 + 6 + 7 + 8 + 9)	-105.5
11. Total received power (1 + 10), dBm	-78.5
12. Receiver noise spectral density (noise temp, 60 K), dBm/Hz	-180.8
13. Carrier power/total power	-23.4
14. Received carrier power (11 + 13), dBm	-101.9
15. Carrier threshold noise bandwidth (18 Hz), dB-Hz	+12.5
Carrier Tracking One-Way	
16. Threshold SNR in 2 B_{LO}	+10.0
17. Threshold carrier power (12 + 15 + 16), dBm	-158.3
18. Performance margin (14-17)	+56.4
19. Data power/total power	-0.7
20. Waveform distortion loss	-0.2
21. Loss through radio frequency system; uplink SNR in 2 B_{LO} one-way downlink SNR in 2 B_{LO}	-0.1
22. Subcarrier demodulation loss	-0.1
23. Bit synchronization loss	-0.2
24. Received data power (11 + 19 + 20 + 21 + 22 + 23)	-79.8
25. Threshold data power (12 + 25a + 25b), dBm	-94.2
a. Threshold E_b/N_o (BER 1×10^{-5}) approx. 9.6 dB	
b. Bit rate 50×10^6 bps, 50.8 (dB-bps) approx. 77.0 dB	
26. Performance margin (24-25)	+14.4

*Unless otherwise specified.

Table 3.3-8. S-Band Command Link
(64 bits per second, digital PSK, 10^{-5} BER, 30-foot station)

Parameter	dB*
Station transmitter power (30 mw), dBm	+14.8
Transmitting circuit loss	-0.5
Transmitting antenna gain (30 ft at 2100 MHz)	+43.1
Transmitting antenna pointing loss	-0.1
Space loss at synchronous altitude (19,300 n mi)	-190.0
Polarization and atmospheric weather losses	-1.7
Receiving antenna gain	0.0
Receiving antenna pointing loss	-0.5
Received total power, dBm	-135.5
Receiving noise power density	
K_T/System noise temperature, 500 K, dBm/Hz	-171.6
Received total power-to-noise power density	+36.1
Carrier Tracking	
Carrier power/total power	-3.5
Received carrier power to noise power density	+32.6
Carrier threshold noise bandwidth $2B_{LO}$ = 18 Hz	+12.5
Received carrier power to noise power ratio	+20.1
Threshold signal-to-noise ratio in $2B_{LO}$	+10.0
Carrier tracking margin	+10.1
Data Performance	
Data power/total power	-3.5
Miscellaneous losses in RF system	-1.5
Received data power, dBm	-139.0
Received data power-to-noise density	+31.1
Threshold E_B/No @ PE = 1×10^{-5}	+9.6
Data rate at 64 bits per second	+18.0
Data performance margin	+13.5

*Unless otherwise specified

Table 3.3-9. Ku-Band Command Link
(64 bits per second, digital PSK, 10^{-5} BER, 30-foot station)

Parameter	dB *
Station transmitter power (437 mw), dBm	+26.4
Transmitting circuit loss	-0.5
Transmitting antenna gain, 30 ft @ 15.30 MHz	+61.0
Transmitting antenna pointing loss	-0.5
Space loss at synchronous altitude, 19,300 n mi	-207.3
Polarization and atmospheric weather losses	-12.6
Receiving antenna gain	0.0
Receiving antenna pointing loss	-0.5
Received total power, dBm	-134.0
Receiving noise power density K_T /System noise temperature, 700 K, dBm/Hz	-170.1
Received total power-to-noise power density	+36.1
Carrier Tracking	
Carrier power/total power	-3.5
Received carrier power-to-noise power density	+32.6
Carrier threshold noise bandwidth, $2B_{LO}$ = 18 Hz	+12.5
Received carrier power-to-noise power ratio	+20.1
Threshold signal-to-noise ratio in $2B_{LO}$	+10.0
Carrier tracking margin	+10.1
Data Performance	
Data power/total power	-3.5
Miscellaneous losses in RF system	-1.5
Received data power, dBm	-139.0
Received data power-to-noise density	+31.1
Threshold E_B/No @ PE = 1×10^{-5}	+9.6
Data rate at 64 bits per second	+18.0
Data performance margin	+13.5

*Unless otherwise specified.

Transmission of housekeeping data at Ku-band through an omni antenna system is also feasible. Ku-band link calculations in Table 3.4-7 indicated that a high-gain antenna (41-1 dB) was required. But those calculations were based upon a 50 Mbps data rate; the housekeeping data rate is only 1 Kbps. For the smaller data rate, an increase in signal margin of 47 dB can be realized. However, the use of an omni antenna system reduces the signal margin of the calculations in Table 3.4-7 by 41.1 dB. Thus, the net effect of using an omni antenna and a 500 mw transmitter to relay 1.0 Kbps data is to increase the overall signal margin by +5.9 dB to a value of +20.3 dB.

Packaging Concept

The four major functions of the TT&C can be conveniently packaged into one replaceable module. The physical characteristics of each subassembly are summarized in Table 3.3-10. The composite packaging arrangement is illustrated in Figure 3.3-14. Complete redundancy is provided in the transmission channels for both the housekeeping and science telemetry.

Table 3.3-10. Physical Parameters

Module	Dimensions (in.)	Weight (lb)
Data Processor		
Power converter	6.6 x 14.0 x 1.2	6.5
Engineering sampler/switch	6.6 x 14.0 x 1.64	4.0
Science sampler/switch	6.6 x 14.0 x 1.64	4.0
Conditioner/control/switch	6.6 x 14.0 x 1.64	4.0
Conditioner/control/switch	6.6 x 14.0 x 1.64	4.0
Multiplexer	6.6 x 14.0 x 1.64	4.0
Multiplexer	6.6 x 14.0 x 1.64	4.0
Volume total	6.6 x 14.0 x 13.5	30.5
Telmetry		
Telemetry modulator power (2)	6.0 x 6.7 x 2.2	6.6
Telemetry modulator logic (2)	6.0 x 6.7 x 1.5	2.8
Command power (2)	6.0 x 6.7 x 2.2	6.6
Command logic (2)	6.0 x 6.7 x 1.5	2.8
Volume total	6.0 x 6.7 x 14.8	18.8
Radio Frequency		
Receiver (2)	4.0 x 5.0 x 1.0	1.5
Modulator/exciter (2) + (2)	4.0 x 5.0 x 3.0	8.0
Power amplifier (2) + (2)	4.0 x 5.0 x 5.0	10.5
Volume total	4.0 x 5.0 x 18.0	20.0

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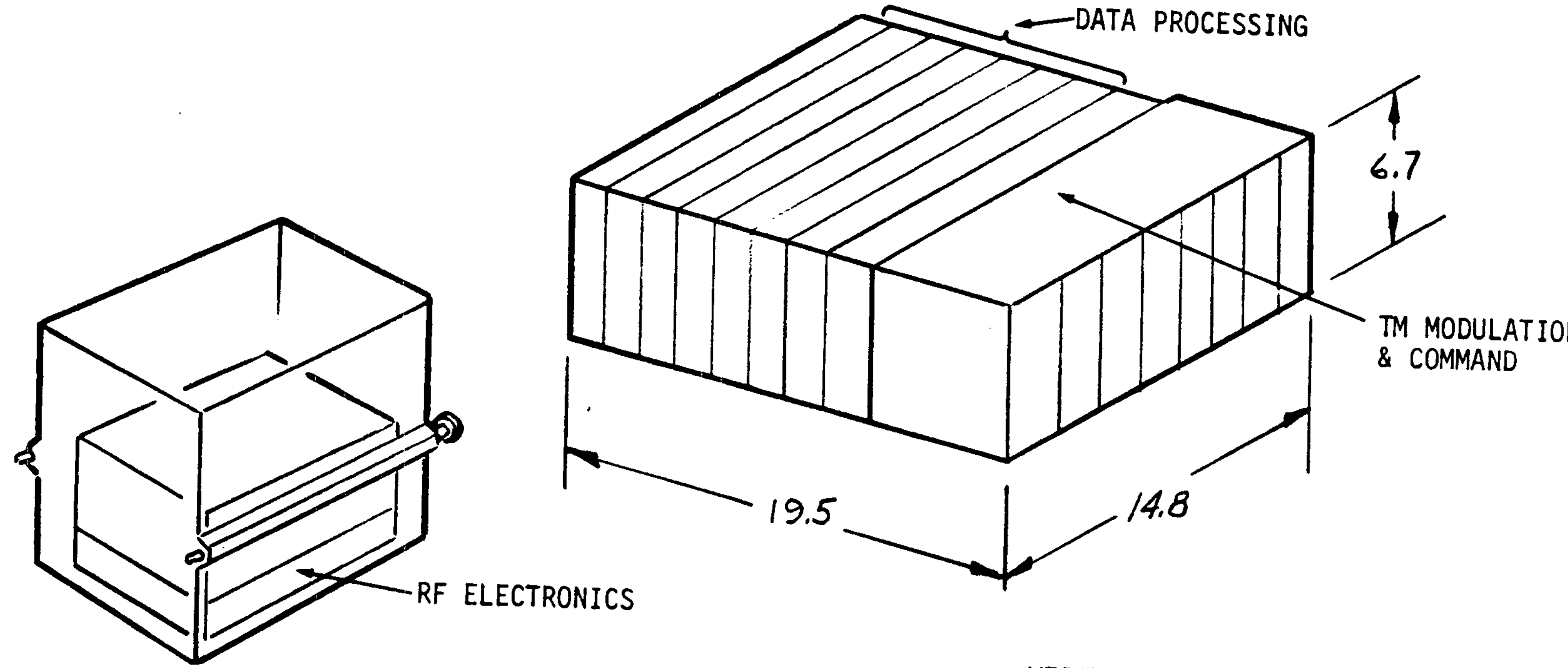

WEIGHT	
DATA PROCESSING	30.5 lb
TM EQUIPMENT	18.8 lb
RF ELECTRONICS	20.0 lb
	59.3 lb

COMPOSITE VOLUME
10.7 x 14 x 19.5

Figure 3.3-14. TT&C Packaging Concept

ATTITUDE STABILIZATION AND CONTROL SUBSYSTEM

The synthesis of the attitude stabilization and control subsystem (ASCS) for geosynchronous platforms is presented in this section of the report. Three primary functions are included in the ASCS: (1) attitude determination, (2) stabilization and control, and (3) propulsion.

Performance requirements of the ASCS are based upon the characteristics of the satellites in the two traffic models of this study, the operational environment at geosynchronous orbit, and the platform configurations developed in Sections 4.0 and 5.0 of this volume of the final report. Alternate ASCS concepts are presented. A preferred concept for three classes of platforms (data relay, earth observation, and astro-physics) as well as a concept applicable for all geosynchronous platforms are defined. Replaceable module packaging for the ASCS is delineated. Three orthogonal reactional wheels are the preferred common stabilization system. Charge-coupled devices that are used as star sensors are preferred for attitude determination. A four-quad hydrazine propulsion system is selected for required maneuvers.

Requirements

System level functional requirements are delineated in Table 3.3-11. Note that north-south stationkeeping is not included as a basic requirement. Only the data relay platforms were identified as potentially requiring this maneuver. Resolution of this requirement is dependent upon an integrated ground terminal-logistics resupply-platform economics trade. However, the selected configuration does have the capability for limited north-south stationkeeping.

Table 3.3-11. ASCS System Requirement

Function	Activity	Requirement
Reliability	Lifetime	Components 10 years Expendables Data relay 10 years Earth observation 7 years Astro-physics 5 years
	Failure criteria	No single failure shall preclude adequate stabilization for tug docking
Torques	Separation	Rate damping
	Angular maneuvers	Three-axis stabilization Three-axis maneuvers
	Disturbance torques	Momentum storage Secular momentum dump
Translation	Initial placement	30 ft/sec delta velocity
	East-west stationkeeping	7 ft/sec/year
	Station change	One at 1 degree/day for TDRSS only--18 ft/sec

ASCS performance requirements are summarized in Table 3.3-12. The listed pointing accuracy and attitude knowledge requirements are beyond the present state of the art of centralized ASCS equipment. The common approach is to utilize the sensor that requires the accuracy as an integral part of the control loop. Therefore, the pointing and stability characteristics of the synthesized ASCS will reflect centralized equipment capability rather than satellite or platform requirements.

Table 3.3-12. Attitude Pointing Accuracy and Maneuver Requirements

Mission	Pointing Reference	Pointing Accuracy	Attitude Knowledge	Rate Stability	Number of Reorientation Maneuvers/ Day	Attitude Maneuver Rates
Data Relay Platform	Local vertical only (on-off null pointing)	0.2 deg	3 $\widehat{\text{minutes}}$	$0.1\ \frac{deg}{sec}$	None	No maneuvering
Earth Observation Platform	Local vertical ±8.5 degrees	2.0 $\widehat{\text{min}}$	1.08 $\widehat{\text{sec}}$	$1.0\ \frac{\widehat{sec}}{sec}$	<240	$0.5\ \frac{\widehat{min}}{sec}$
Astro-physics Platform	Inertial - 4π steradian	1.0$\widehat{\text{min}}$	≈1 $\widehat{\text{sec}}$	$0.05\ \frac{\widehat{sec}}{sec}$	6	$0.1\ \frac{\widehat{deg}}{sec}$
Satisfies all missions	4π steradian	1.8 $\widehat{\text{min}}$	0.1 $\widehat{\text{sec}}$	$0.05\ \frac{\widehat{sec}}{sec}$	240	$0.1\ \frac{\widehat{deg}}{sec}$

The integrated set of control system requirements of the earth observation platform are the most demanding. Because of the inclusion of a telescope in this platform, a scanning maneuver is required to view the entire global region within the field of view of that platform. The characteristics of the assumed scan pattern are illustrated in Figure 3.3-15. Each frame is scanned by the sensor in approximately four minutes. Each successive westward frame must be scanned within a 5.61-minute interval to preserve equal solar illumination between frames. Although the maneuver rate requirements are not high, the torque requirements to maneuver the sawtooth attitude profile are very demanding for conventional reaction wheel systems.

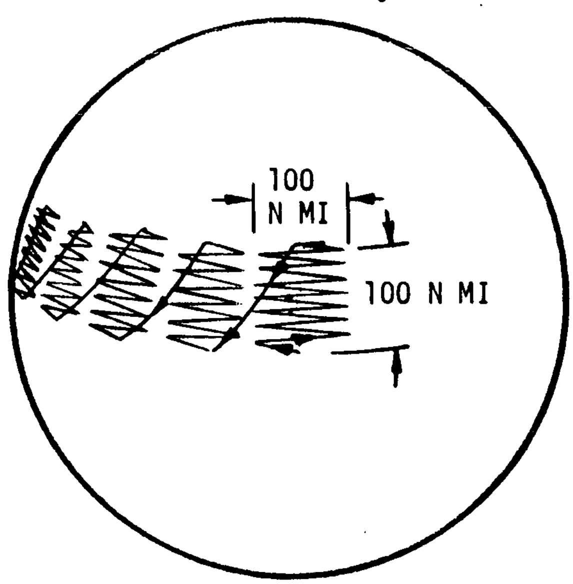

Figure 3.3-15. Earth Observation Platform Scan Pattern

Disturbance Torque Environment

The dominant disturbance torque acting upon geosynchronous platforms is due to solar pressure. This torque affects the sizing of the momentum storage/transfer assemblies and the required propellant expendables. Based upon the level of platform definition attained in this study, estimates of the solar pressure disturbance torques and the associated angular momentum transferred to the platform are presented in Table 3.3-13.

Table 3.3-13. Summary of Momentum Requirement Due to Solar Pressure Torque

Platform	Maximum Cyclical Momentum Variation per Orbit (~ft-lb-sec)			Secular Growth per Orbit (~ft-lb-sec/day)	Orientation
	X	Y	Z		
Data Relay	0.8	2	0.4	0.98	Local vertical
Earth Observation	4	10	2	6.5	Local vertical
Astro-Physics	Not Applicable			9.75	Worst case inertial

Internal disturbance torques such as rotating machinery, antenna gimbaling, sensor gimbaling, etc., could also directly affect the momentum storage system. Only the TDRS was identified as having a potential internal disturbance torque (antenna gimbaling). However, the magnitude of the disturbance was not a significant driver in the ASCS sizing.

System Synthesis Trades

The development of new ASCS techniques and improvements in component technology are taking place at a very high rate. These developments could invalidate current trade study results well before platform designs must be frozen. For this reason, the purpose of the ASCS synthesis trade studies is to select viable representative system approaches. ASCS systems are synthesized for the data relay, earth observation, and astro-physics platforms in order to determine if there are penalties in using a common system for all platforms.

The key elements of the ASCS are the torque source elements [RCS and momentum storage/transfer system (MSTS)] and attitude reference determination system (ARS). The trade trees illustrating possible mechanization techniques are given in Figures 3.3-16 and 3.3-17. In the MSTS trades the momentum approach is attractive for the earth pointing platforms. The two-wheel "T" configuration is selected for the data relay platform, and the two-degree-of-freedom gimbaled reaction wheel is selected for the earth observation platform. Three orthogonal momentum wheels are selected for the astro-physics platform.

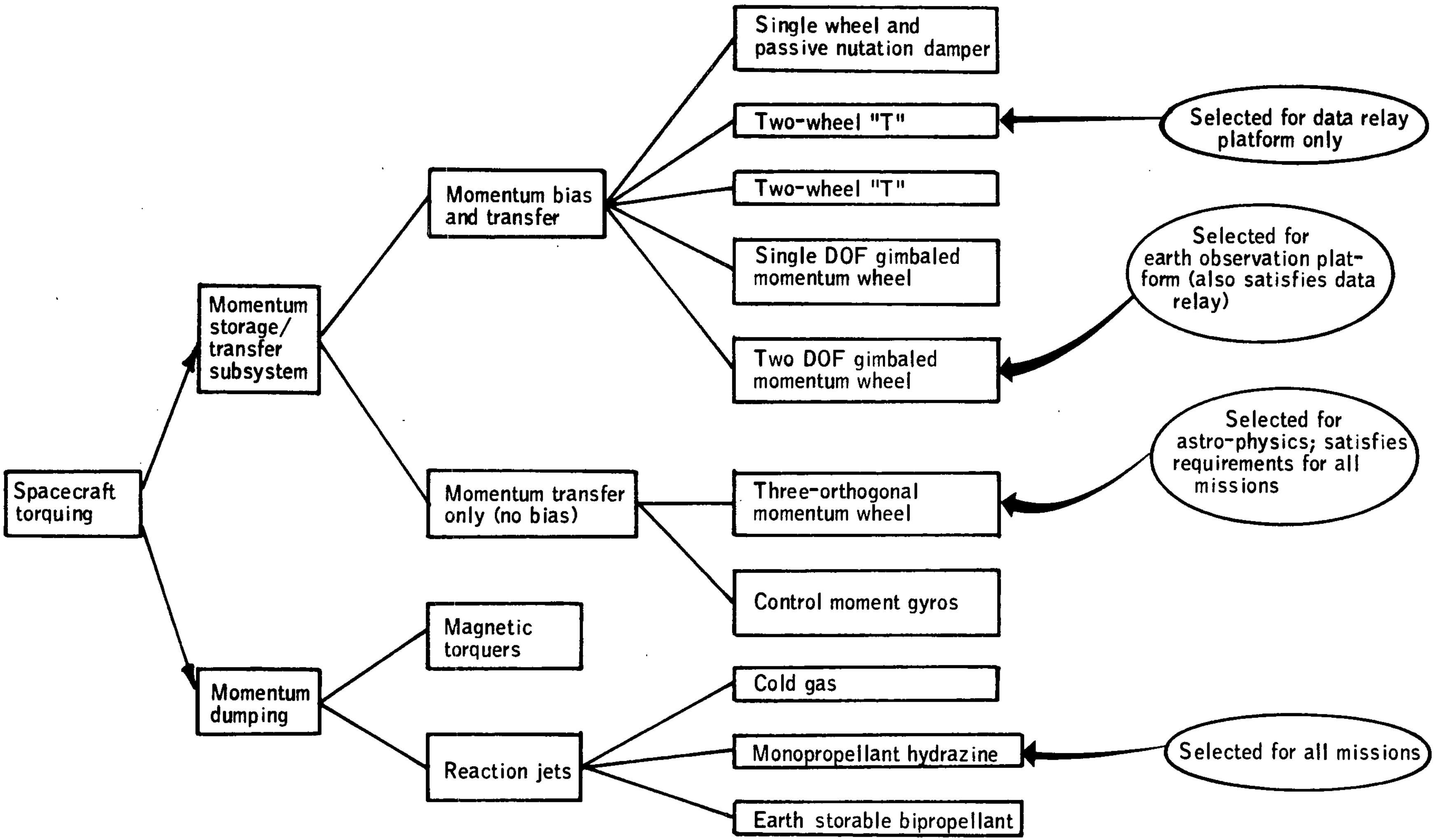

Figure 3.3-16. Attitude Control Torque Source Trade Tree

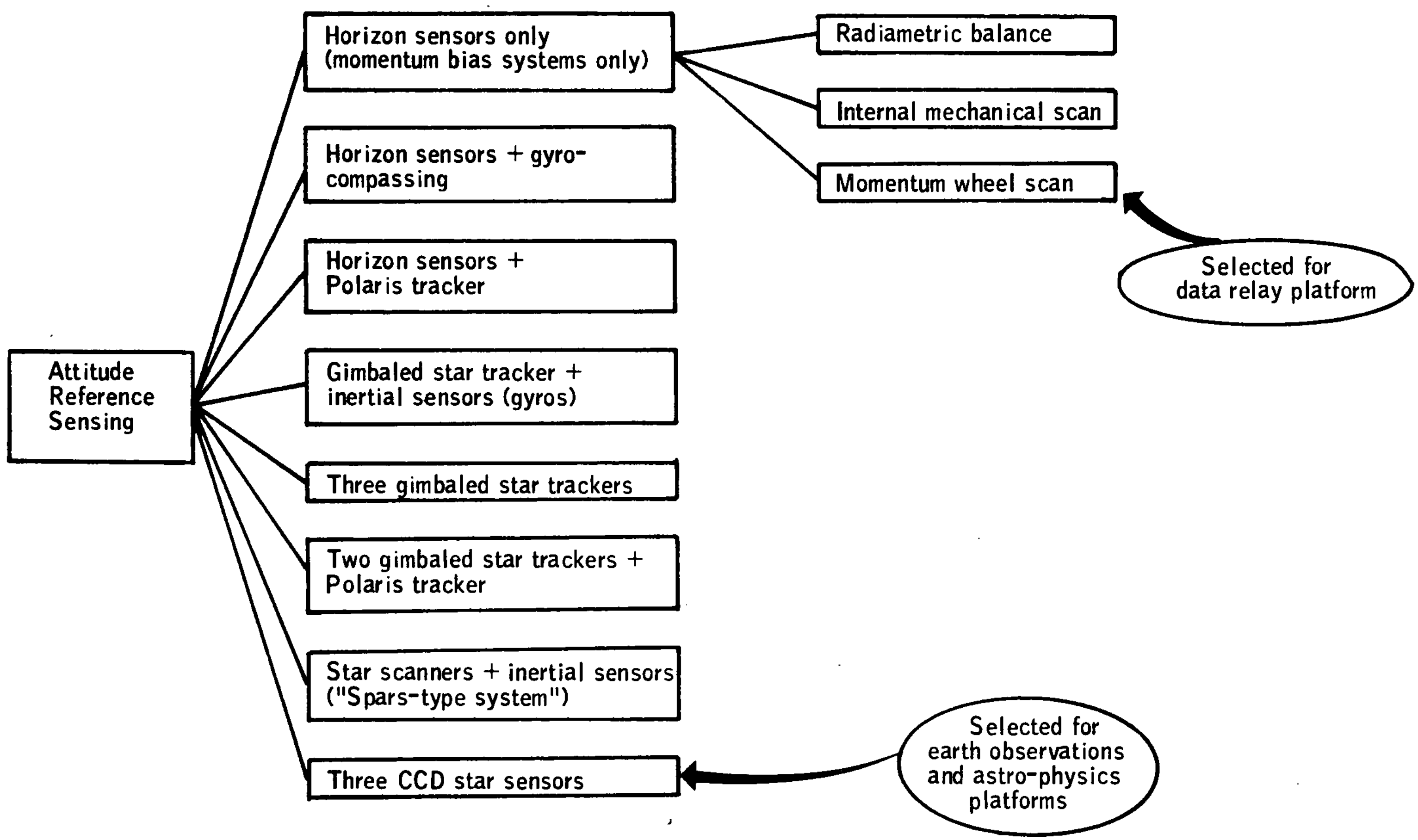

Figure 3.3-17. Attitude Reference System Trade Tree

For momentum dumping the reaction jet propellant requirements are normally quite small. The monopropellant hydrazine system is selected over cold gas or earth storable liquid propellants on the basis of system weight (see Figure 3.3-18), low cost, well-developed existing technology, and ease of propellant handling and servicing.

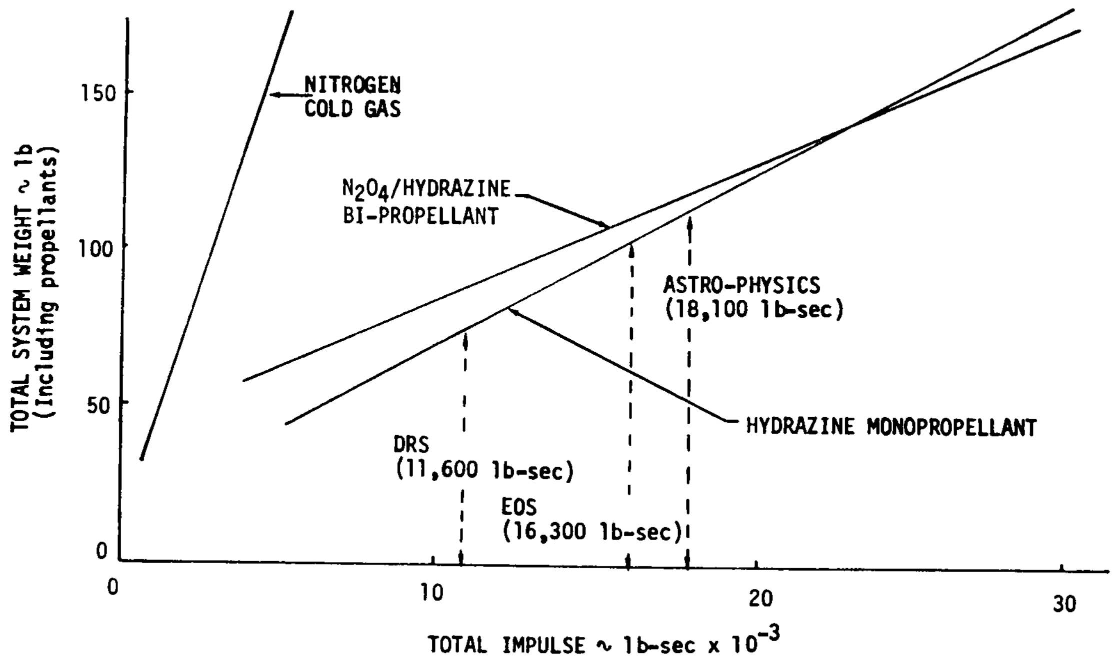

Figure 3.3-18. Weight Comparison of Candidate Propulsion Systems

For attitude determination the available technology provides sun, stellar, horizon,and inertial sensing instruments. Alternative sensor combinations to perform three-axis attitude determination were given in Figure 3.3-17. Sun sensors are excluded due to insufficient accuracy and earth occultation problems. Platform control necessitates a moderate bandpass requirement and many of the combinations require an inertial reference system (gyros) for this purpose. Use of the charge-coupled device (CCD) star sensor telescopes was selected for the earth observation and astro-physics platforms. Only 50 milliseconds are required to perform the attitude determination and thus no inertial reference system is required. The horizon sensor is selected for the data relay platform. Lower accuracy requirements coupled with the momentum bias two-wheel "T" concept also preclude a requirement for an inertial reference system.

All selections were based primarily on minimum estimated weight and cost for the candidate concepts.

A summary of the estimated physical characteristics and relative cost of the ASCS systems for the three types of platform are presented in Table 3.3-14. Note that the data relay platform concept results in a substantial cost and weight savings over the astro-physics system which will handle all platform requirements. The earth observation platform concept offers a moderate weight savings over the astro-physics system but the cost differences are small.

Table 3.3-14. Dedicated ASCS Comparison

Mission	System	Weight (~lb)	Average Power (~watts)	Peak Power (~watts)	Estimated Relative Cost (%)
Data Relay	"T" momentum bias and wheel-scanned horizon sensing	66	45	236	43.7
Earth Observation*	Two-degree-of-freedom gimbaled momentum wheel and strapdown star sensors	105	62	296	96.5
Astro-Physics**	Three orthogonal reaction wheels and strapdown star sensors	149	80	406	100.0

*System also accommodates data relay requirements
**Compatible with requirements of all missions

System Descriptions

Three separate ASCS configurations were synthesized for the requirements of the three types of geosynchronous platforms. This is done to establish any penalties that might be inherent in using a common system for all platforms. The operation of the ASCS is illustrated in the flow diagram of Figure 3.3-19. A momentum storage/transfer system (MSTS) is included to provide for maneuvering and countering the secular disturbance torques without propellant expenditure. The selected MSTS configurations are illustrated in Figure 3.3-20. No redundancy is included in the MSTS since the "fail-safe" redundancy criteria require only that the system be able to stabilize itself for tug docking and subsequent servicing. This can be accomplished with the reaction jets which are redundant.

A component summary for the three systems is presented in Table 3.3-15. The DRS system features a "T" momentum bias configuration in which the earth horizon sensor is scanned by the momentum bias wheel. It may be observed from Table 3.3-12 that the performance requirements for the data relay platform are not very demanding and, hence, it is no surprise that the system is considerably less expensive and lighter than the other systems. The earth observation platform has the most severe disturbance torque environment and performance requirements. However, its near local vertical orientation permits the use of another momentum bias approach, the two-degree-of-freedom gimbaled reaction wheel, and hence results in a lower system weight than the inertially stabilized astro-physics platform. The astro-physics platform system utilizes a conventional momentum wheel triad configuration.

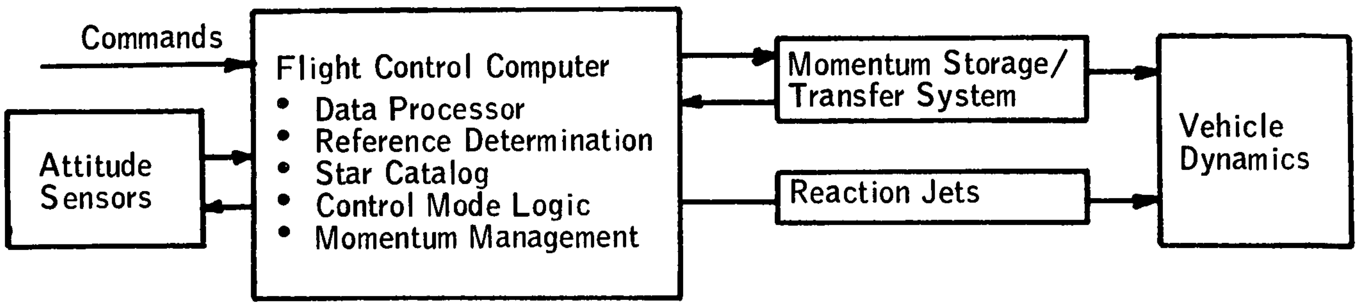

Figure 3.3-19. Attitude Stabilization and Control System

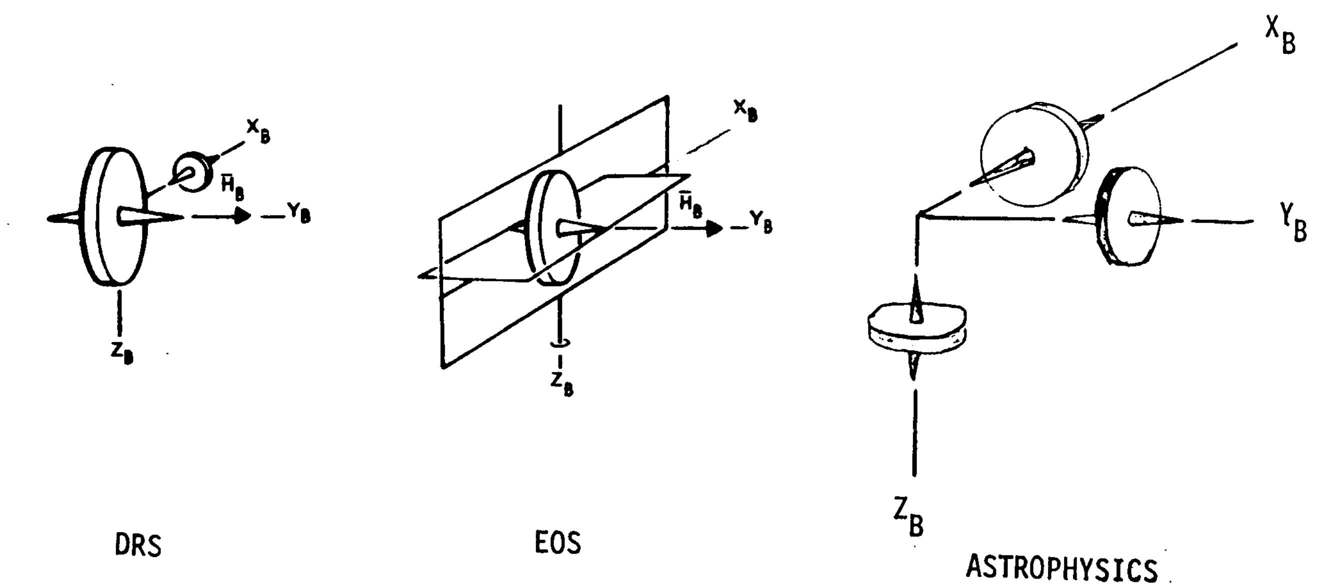

Figure 3.3-20. Momentum Storage/Transfer Configurations

Table 3.3-15. Summary of ASCS Characteristics

Platform	Component	No. per S/C	Weight (lb)	Power (~watts)		Maximum Dimensions per Unit (in.)
				Average	Peak	
Data Relay	Pitch axis momentum bias wheel (Sperry Model 45Q)	1	25	10	100	15 dia x 10
	Yaw axis momentum wheel (Bendix)	1	10	5	50	10 dia x 4.25
	Horizon sensor ("X" scanner by Ithaco, scanned by pitch wheel)	1	5	20	-	6 dia x 5
	Flight control computer and electronics	1	26	30	86	18 x 10 x 5
	Totals		66	65	236	
Earth Observation	Two-degree-of-freedom gimbaled momentum wheel (Sperry Model 75)	1	35	12	190	18 dia x 18
	CCD star sensor telescopes and sunshields	3	30	20	-	5 dia x 10 and 10 dia x 8
	Flight control computer and electronics	1	40	50	106	18 x 22 x 10
	Totals		105	82	296	
Astro-Physics	Momentum wheels, three orthogonal (Sperry Model 45Q)	3	75	30	300	15 dia x 10
	CCD star sensor telescopes and sunshields	3	30	20	-	5 dia x 10 and 10 dia x 8
	Flight control computer and electronics	1	44	50	106	18 x 22 x 10
	Totals		149	100	406	

The attitude reference sensors selected for the earth observation and astro-physics platforms are the CCD star sensor telescopes. The sensor detector consists of a charge-coupled device (CCD) solid-state mosaic array which is digitally scanned. The resulting outputs are compared and correlated with star field catalog data in the flight control computer. The telescope field of view (approximately 10 degrees) is sufficient to always permit the observation of at least one star greater than 8.5 magnitude. Two telescopes provide sufficient data for a complete attitude reference determination. Three telescopes are used in a near orthogonal configuration so that complete attitude reference data are available when one telescope is interfered with by the sun or earth. A small wide-field-of-view sensor (approximately 40 degrees) is included in each star telescope package and serves two purposes. It facilitates the initial attitude reference acquisition (star identification) and serves as a backup sensor in the event of the failure of a primary telescope. The star sensor mosaic scanning, star identification,and attitude reference computations can all be performed in less than 50 milliseconds. The CCD star sensor concept therefore provides an essentially continuous (wide bandpass) system and eliminates the requirement for an inertial reference package (gyros) for attitude reference between star observations.

Of all the ASCS concepts evaluated, the three-orthogonal reaction wheel/ charge-coupled device system is considered the preferred approach on a commonality basis. It can be used for all three types of platforms. The differences between this universal concept and the "optimized" approach for earth observation platforms are not sufficient to warrant separate developments. The significant difference between the universal approach and the data relay platform approach is the cost. Two factors must be considered in evaluating the relative costs of the concepts: development status and equipment installation/ field-of-view interferences.

Horizon scanners are well developed off-the-shelf hardware and the costs reflect this development status. The charge-coupled device (CCD) concept is an emerging technology. Prototype components have been developed and system feasibility studies have been conducted; but an integrated sensor package has not been developed. The CCD concept is completely solid state and requires no mechanical scanning. A substantial increase in reliability is anticipated. It is basically a digital imaging sensor. The CCD concept may see application in the commercial television industry by the 1980 time frame. If development progresses as some forecasts predict, the CCD concept could be cost competitive with the horizon scanners.

Placement of the horizon scanner in the platform must of course permit line of sight of the limb of the earth. The CCD concept can operate over 4π steradians except within about 35 degrees of the sun. Also, in order to avoid extraneous reflections a cone of clearance from other spacecraft appendages of about 25 degrees is required.

The platform concept of a common support module coupled with a mission equipment module results in certain design problems in placing the horizon scanner. Three options for integration of horizon scanners into data relay platforms are: (1) place the common support module on the earth side of the platform, (2) extend the horizon scanner on a boom from the common support module, and (3) incorporate the horizon scanner in the mission equipment module on the earth side of the platform.

All three options are undesirable. Placing the common support module on the earth side of the data relay platform severely complicates the interface between the two equipment rings. All RF connections between antennas and transponder would have to be routed through the common support module. Extension of the horizon scanner on a boom increases the mechanization complexity and reduces system reliability. Inclusion of the horizon sensor in the mission equipment ring perhaps is the least objectionable, but certainly violates the concept of a common support module. Also, one of the cost advantages of the "T" wheel-horizon sensor concept is the integral packaging of its components.

The potentially large inventory of data relay platforms may warrant a separate development of the "T" wheel horizon scanner ASCS concept. However, for purposes of this study the three-orthogonal reaction wheel-CCD ASCS concept that is preferred for the Astro-Physics platform will be used as the baseline concept.

Auxiliary Propulsion System

The propellant requirements for the selected hydrazine system for the three types of platforms are presented in Table 3.3-16, and are based on the

Table 3.3-16. Propellant Requirements

Characteristics \ Platform	Data Relay	Earth Observations	Astro-Physics
Initial weight (lb)	3000	5500	7500
Non-serviced mission duration (yr)	10	7	5
Propellant requirements (lb)			
. Initial placement correction (30 ft/sec)	12.7	23.3	31.8
. Station change (TDRS only) (18 ft/sec)	7.6	--	--
. East-west stationkeeping (7 ft/sec/yr)	29.6	38.0	37.1
. Momentum dump (worst case secular torque)	2.7	12.6	13.5
Total propellant required (lb)	52.6	73.9	82.4

requirements given in Table 3.3-11. It may be seen that two fully tanked propulsion modules (44 pounds of propellant each), providing a total propellant capacity of 88 pounds, can sustain all of the platforms for a relatively long mission without additional propellant servicing. Two propulsion modules are therefore considered to be adequate. An appropriate reaction jet configuration for the two-module system is illustrated in Figure 3.3-21. The arrangement provides for torques about all three axes and for delta-V maneuvers. Redundant jets are included to accommodate the failure of at least any single jet. Somewhat more limited operation is possible with a tank failure in that some cases may require a 180-degree yaw reorientation to thrust in the proper direction for stationkeeping.

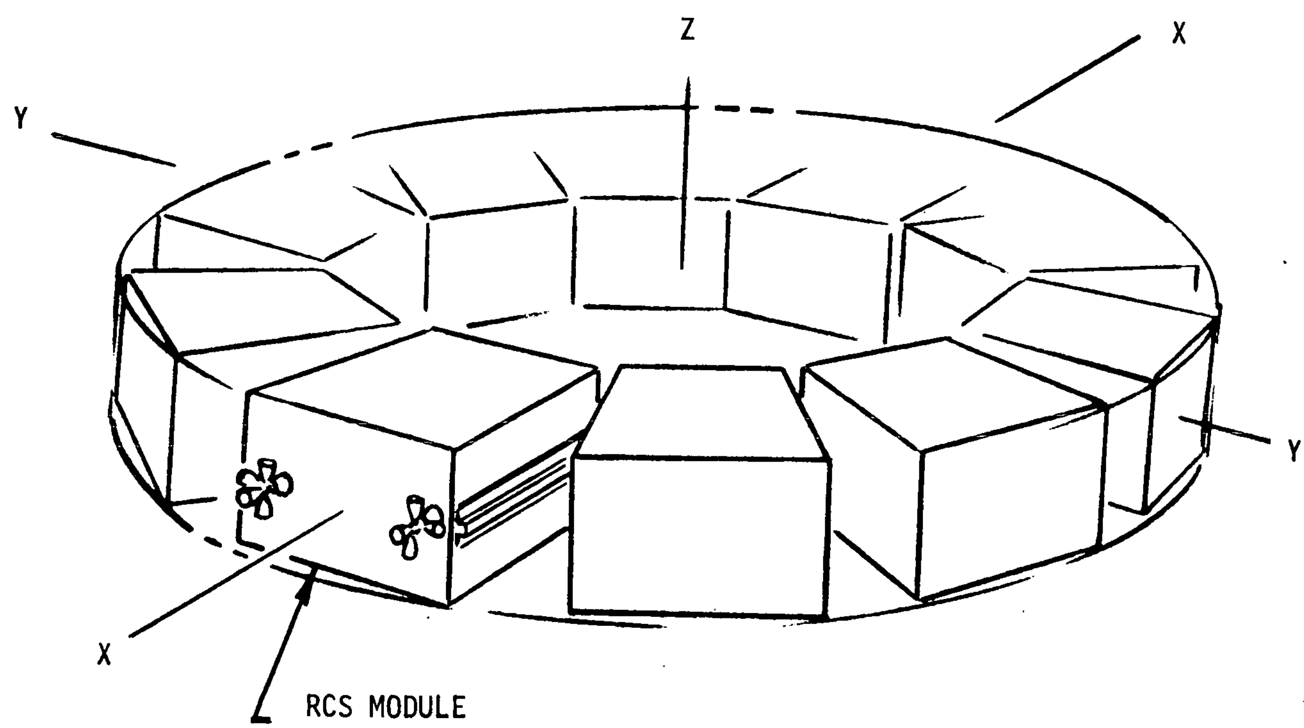

Figure 3.3-21. Reaction Jet Configuration
(Two Modules--16 Engines)

An option exists for the addition of two additional propulsion modules. The basic configuration of the common support module will accommodate 12 standard equipment assemblies. The basic support subsystems require only 10 assemblies. A potential requirement for north-south stationkeeping of data relay platforms has been suggested. Combining the basic two propulsion units into an additional two units would provide 176 pounds of propellant for total operations including north-south stationkeeping. North-south stationkeeping requires approximately 150 ft/sec/year delta V. The resultant nominal mission duration of a data relay platform without propellant replenishment could be as long as 2.5 years.

The addition of two propellant modules to the basic common support module is not a platform concept driver. It permits consideration of north-south stationkeeping or simply increased redundancy of the propulsion system. Neither the cost nor the weight of the two additional units will significantly affect programmatic comparisons. Therefore, for purposes of this study, it will be assumed that the common support module has four propulsion units.

Figure 3.3-22 illustrates a four-quad propulsion system. Each engine is rated at one-pound thrust with an I_{sp} of 220. The revised engine cluster for the four-quad concept facilitates shirtsleeve servicing. A concept for shirtsleeve servicing of the quads is also shown in Figure 3.3-22. A cover that hinges on the common support module structure will, when closed, effectively provide a pressure seal around a structural cutout through which the jets protrude.

ASCS Packaging Concepts

The ASCS may be packaged modularly as indicated below:

- RCS - four modules
- Momentum storage/transfer system, drive amplifier electronics and wheel speed control logic - one module
- Flight control computer and electronics - subsection of data processing module
- Star sensor telescopes (three CCD's) - one module including electronics

Figure 3.3-23 illustrates a typical RCS module and sizing criteria. For a 44-pound propellant capacity the maximum dimension, excluding the thrusters, is 20 inches. The three-orthogonal wheel momentum storage module is illustrated in Figure 3.3-24. The flight control computer is depicted in Figure 3.3-25. The CCD assembly is illustrated in Figure 3.3-26. All assemblies can be accommodated in a standardized 24-inch by 20-inch by 24-inch module.

C2

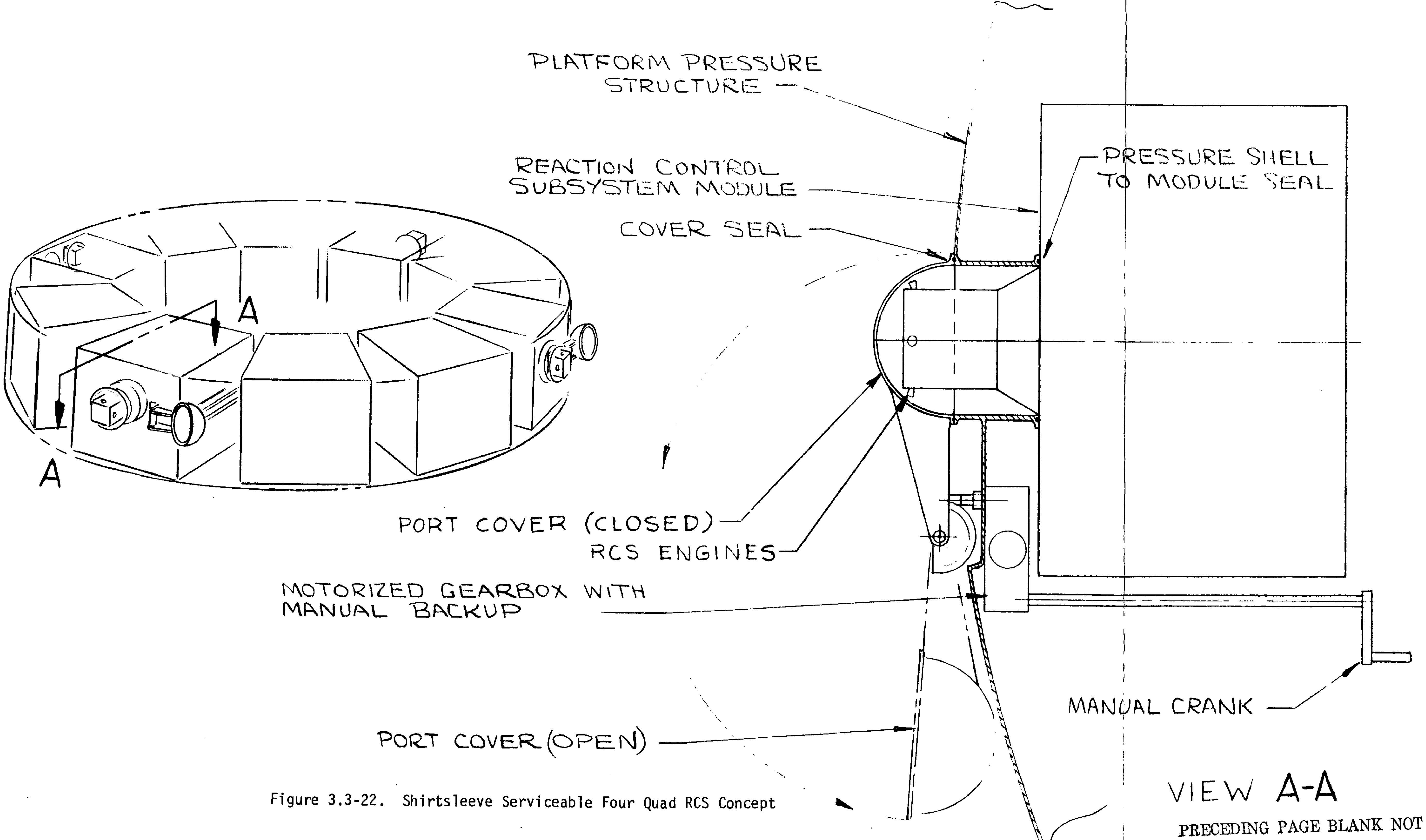

Figure 3.3-22. Shirtsleeve Serviceable Four Quad RCS Concept

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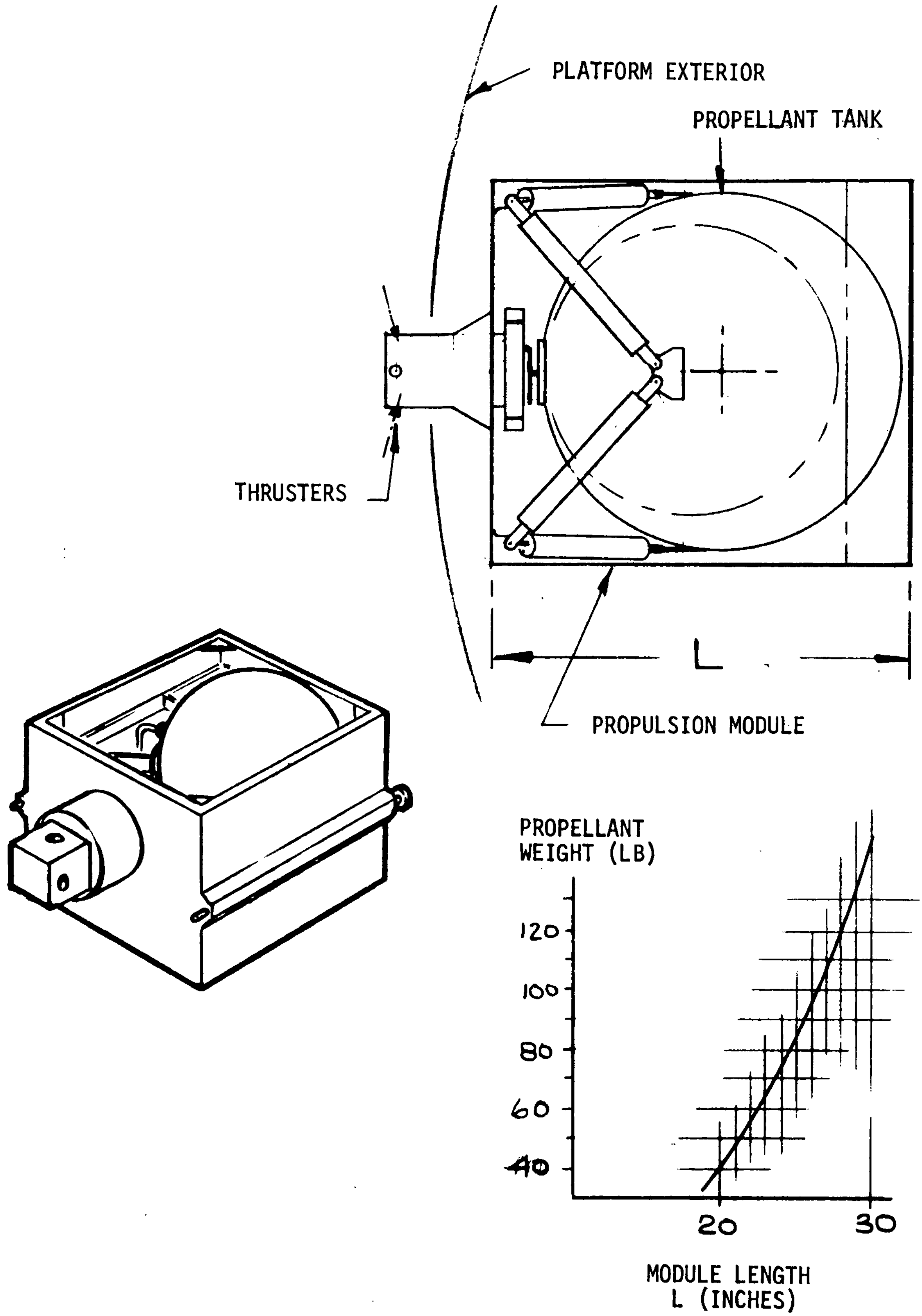

Figure 3.3-23. RCS Propellant Module

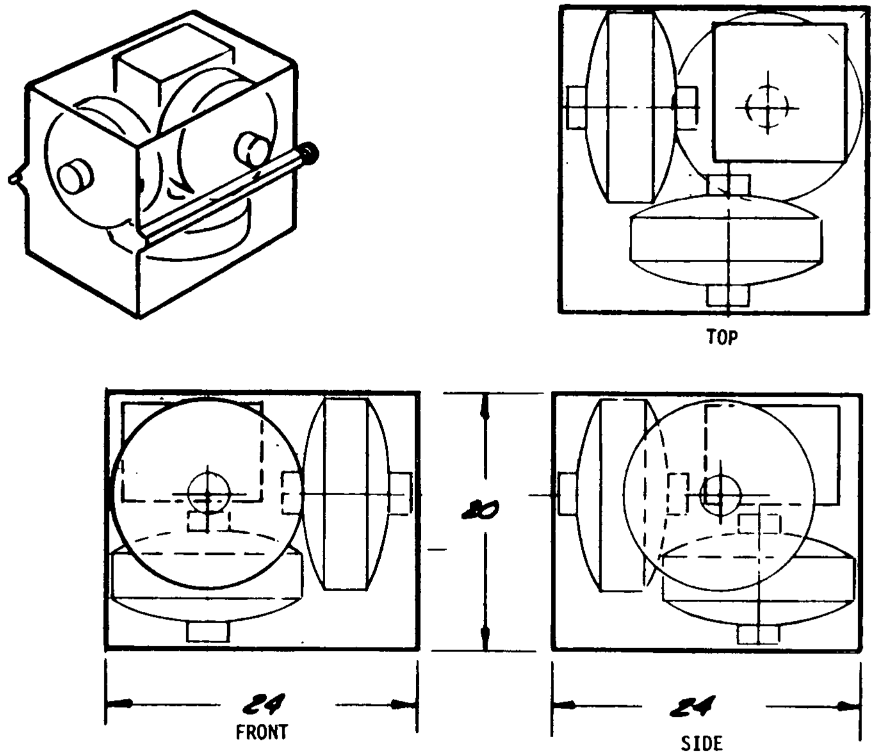

Figure 3.3-24. Reaction Wheel Module

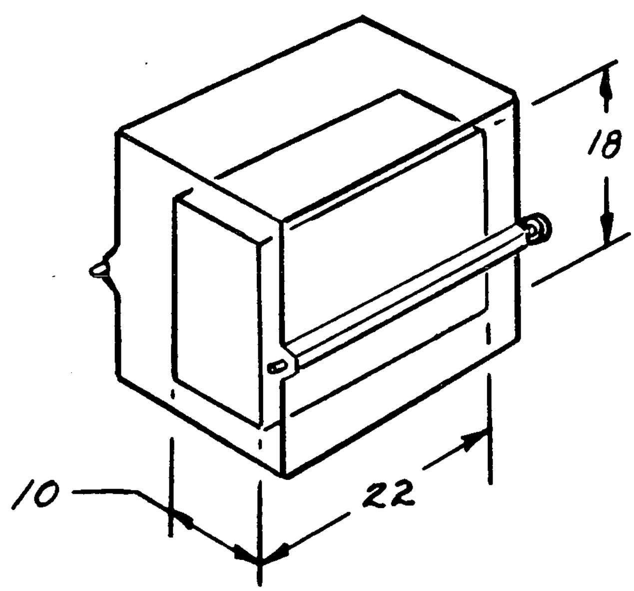

Figure 3.3-25. Flight Control Computer

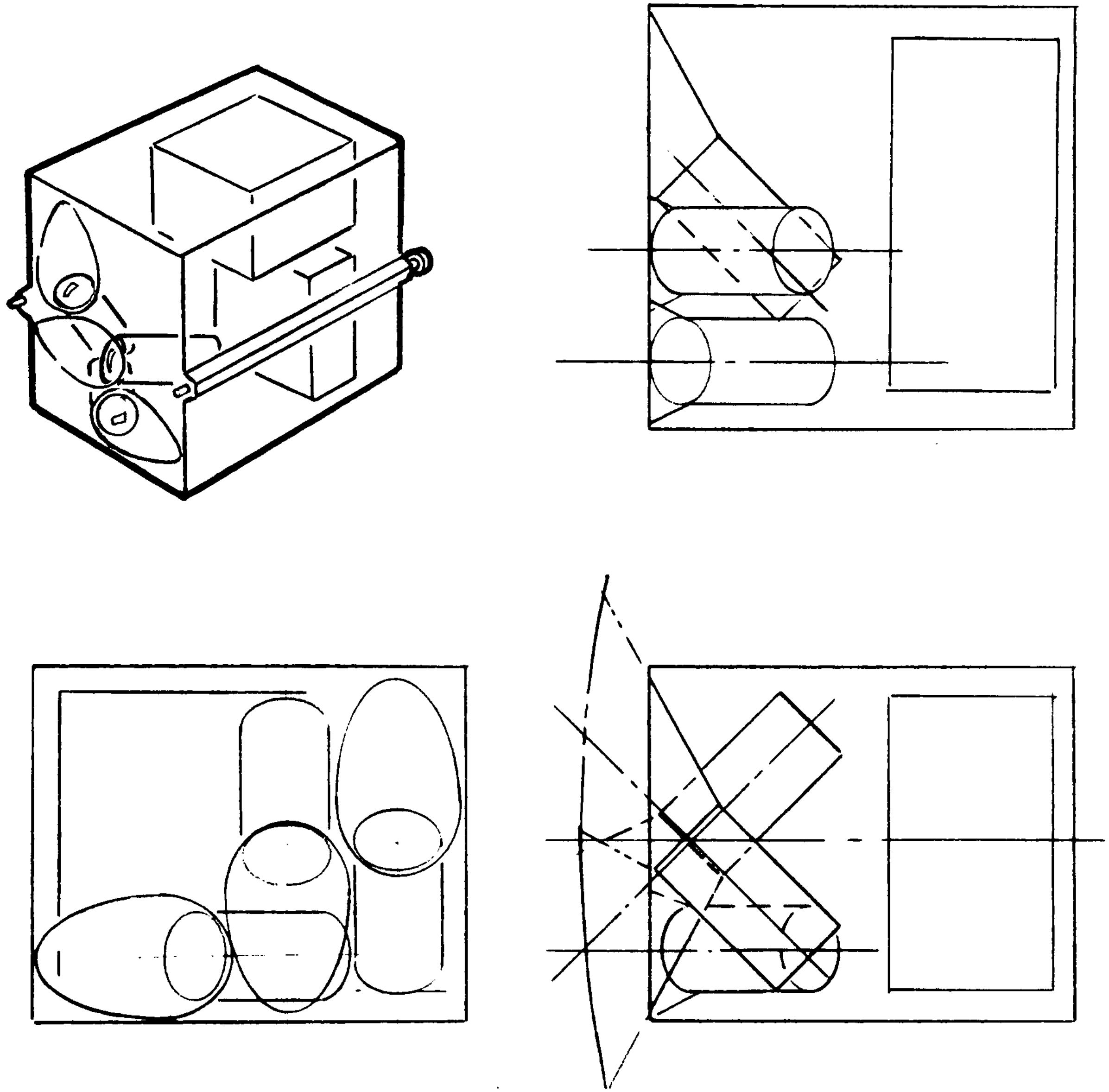

Figure 3.3-26. Charged Coupled Sensor Module

THERMAL CONTROL SUBSYSTEM

The function of the thermal control subsystem (TCS) of the geosynchronous platform is to maintain equipment temperatures within allowable temperature range during delivery, on-orbit operation, servicing operations, and non-operating (turndown) periods. In this section the operating requirements of the TCS are developed, the thermal constraints of the equipment are defined, the impact of the quiescent environments are identified, alternate TCS concepts are discussed, and the preferred concept for the evolutionary geosynchronous platform is defined.

The majority of the data pertaining to the TCS is in a parametric format. As the mission equipment becomes more definitized and optimized in subsequent studies, the TCS data presented herein can readily be assessed for use in the definition of an integrated platform design. Comparisons are made between manned serviceable and auto-remote serviceable platforms as well as the impact on a singular design that can evolve from auto-remote to manned servicing. The application of louvers, heat pipes, optical surface reflectors,and heaters is discussed.

The preferred TCS concept in this study is a heat pipe-radiator concept with a conducting grease mating interface between replaceable modules and the coldplate structure of the platform for modules that dissipate more than 50 watts/ft^2. The requirement for a concept that would accommodate both auto-remote and shirtsleeve servicing was the primary reason for the selected TCS concept. Shirtsleeve maintenance requires equipment modules to be encased within an insulated pressure hull. The pressure hull forces the TCS to have a distinct heat transport assembly to transfer heat from the internal equipment through the pressure hull to an outside surface that can radiate to space. Heat pipes were selected because of their passive and thus highly reliable nature.

If only auto-remote servicing were required, an open truss-type structure of the platform would be preferred from a thermal control standpoint. Each replaceable module could be thermally controlled independently of all other modules, and the majority, if not all, of the modules could be adequately thermally controlled with only optical surface reflection.

Requirements

Design requirements for the TCS include equipment operating temperatures, internal heat dissipation rates, and definition of the external environment.

Equipment Operating Temperatures

Typical temperature requirements for the equipment of a geosynchronous platform are shown in Table 3.3-17. For the open-truss construction the design temperature limits are defined by the lowest maximum and highest minimum temperatures of the components included in a specific module. The design for shirtsleeve maintenance requires the same type of logic applied to all components located inside the pressure hull.

Table 3.3-17. Temperature Requirements of Typical Geosynchronous Platform Equipment

Subsystem/Component	Allowable Temperature (°F)			
	Operational		Survival	
	Maximum	Minimum	Maximum	Minimum
S&C				
Reaction wheels	185	165	190	120
Stellar sensors	100	0	120	-65
Electronics	100	0	120	-65
Propulsion				
Tanks	100	0	120	-10
Lines and valves	110	40	145	-10
Thrusters	145	5	-	0
EPS				
Batteries	75	30	100	10
Control and logic	150	35	170	0
DC voltage regulators	150	35	170	0
Battery charge regulators	150	35	170	0
Load regulators	160	20	180	-20
Solar cell array	150	-150	240	-200
Struts	260	-225	-	-
Drives	260	-70	320	-75
Communications				
Transponders (C- and K-band)	105	30	122	-65
Antennas	150	-150	240	-200
TT&C				
Logic	105	30	122	-65
Transceiver	105	30	122	-65
Data processing	100	0	120	-65
Thermal Control System				
Insulation	250	-200	250	-200
Louvers	130	-50	150	-65
Coatings	200	-200	250	-300
Structure				
Internal walls when occupied	105	58	120	35

Examining the "Operational" columns of Table 3.3-17, the lowest maximum temperature is defined by the batteries while the highest minimum is established by the reaction wheel and the gyro. Temperature maintenance for the latter two components must be accomplished by heaters to guarantee the required elevated temperatures. The next highest minimum is established by the propellant lines. If the thermal control concept can maintain equipment temperatures between 40 F and 75 F, none of the equipment (excluding the reaction wheels and gyros) will be exposed to an unacceptable environment. Since propellant lines are critical to vehicle survival, heaters might also be placed on these lines to provide design margin. In keeping with this design margin approach the final design temperature requirements selected are 40 F and 70 F. Additional margin can be obtained by equipment location. The propellant lines can be routed near heat dissipating equipment and the batteries and solar sensors can be located in colder areas such as those not exposed to radiant heating.

Manned operation has a significant effect on temperature requirements. During manned attendance,habitable volumes will have to be maintained above 58 F to prevent condensation on internal surfaces. Special design approaches such as insulating cold components within the habitable environment, or heating the air which circulates in the volume, will eliminate the potential for condensation. Touch temperatures must also be limited to a maximum of 105 F.

Internal Heat Dissipation

Table 3.3-18 presents heat loads for representative replaceable modules of three typical geosynchronous observation platforms. The rejection requirement for the shirtsleeve-maintained vehicle is the sum of all the heat loads located in the support module for a given mission. In general, the loads vary from 10 to 500 watts per replaceable module. Table 3.3-19 presents the heat loads for the modules of the common support module.

Internal heat generation variations occur when equipment duty cycles vary. For example, heat dissipation on the Tracking and Data Relay Satellite varied during orbit daylight and orbit darkness between 89 watts/hour and 61 watts/hour, respectively; a turndown ratio of 1.5 to 1. Wider or narrower ratios may occur for a platform depending on its design.

External Environment

The external environment varies with the mission phase. During the first mission phase the geosynchronous platform is delivered to low-earth orbit. Initially, the platform will ride in the shuttle cargo bay and is exposed to the environment caused by the interaction of the tug, the cargo bay and the payload. The payload experiences a cold environment caused by the cryogenically stored propellant of the tug and the lack of internal heat generation within the geosynchronous platform. Since the shuttle doors are opened soon after launch and the cargo bay is purged with warm nitrogen gas until launch, the payload should not experience significant cooldown from the tug.

Table 3.3-18. Typical Geosynchronous Observation Platform Heat Loads

Mission	Module	Dissipation Rate (Watts)
Earth Observation and Physics	Imaging camera	21
	Spectral polimeter	20
	Absorption spectrometer	15
	Cloud camera	15
	Sferics detector	6
	Data collection system	8
	Microwave scanner and radiometer	255
	Radiometer/scatterometer	150
	Multispectral TV	200
	IR camera	24
	Aeronomy spectrometer (2)	260
	Scan platform sensor electronics	482
	Scan platform sensor electronics	473
	Ion temperature & density sensor	12
	Cosmic dust detector	10
	Particle detector	50
	VLF electronics	120
	Flux gate magnetometer	20
	IR camera	24
	Multispectral TV	200
	Multispectral IR scanner	130
	Multispectral spectrometer	170
	Multispectral radiometer	20
	Total	2585
High-Energy Physics	Magnetic spectrometer	500
	Absorbtion shower counter	45
	Gamma ray detector	10
	Cosmic ray detector	15
	Cerenkov counter	10
	Total	580
Plasma Physics	Electron accelerometer	130
	Mass spectrometer	35
	Interferometer/spectrometer	457
	Particle detector	50
	Cosmic dust detector	10
	Hemispherical analyzer	5
	LF transmitter	100
	EUV spectrometer	16
	VLF receiver	20
	Magnetometer	20
	Quadrupole mass spectrometer	18
	Ion/electron monitor	14
	Total	885

Table 3.3-19. Common Support Module Heat Loads

Module	Dissipation Rate (Watts)
Battery pack	20
Solar array drive	2
Power conditioning	20
Reaction wheels	30
Star tracker	20
Data processor	28
Power distribution	200
Total	320

The next mission phases occur when the tug and the platform are deployed and initial activation is undertaken for checkout, and then the platform is boosted to geosynchronous orbit. Modules not operating might be facing deep space and could cool down significantly. This problem is, however, similar to the duty cycle problem expected during normal geosynchronous operations. The non-operational period during activation is not expected to be long; therefore, significant thermal problems are not envisioned during this mission phase that would not be solved by the subsystem designs required by geosynchronous operation. The same is true as the platform is boosted to geosynchronous orbit by the tug.

The external environment varies with orbit position when the platform achieves its operational phase in geosynchronous orbit. As the geosynchronous platform orbits the earth it will occasionally be shadowed by the earth. Figure 3.3-27 shows the frequency and duration of this shadowing. Solar heating will vary from 130 watts/ft^2 in sunlight and zero in the shade. Albedo and earthshine effects are negligible at geosynchronous altitudes.

Thermal Control of Open-Truss Platform Design

Auto-remote maintenance allows the platform design to take on an open-truss configuration. Surfaces of the modules are then exposed directly to the space environment. Each module can be thermally controlled independent of the other modules. Heaters, louvers, and thermal control coatings can provide a very reliable TCS concept. The application of these elements is a function of the ratio of high to low heat loads for the module and the minimum and maximum allowable operating temperatures.

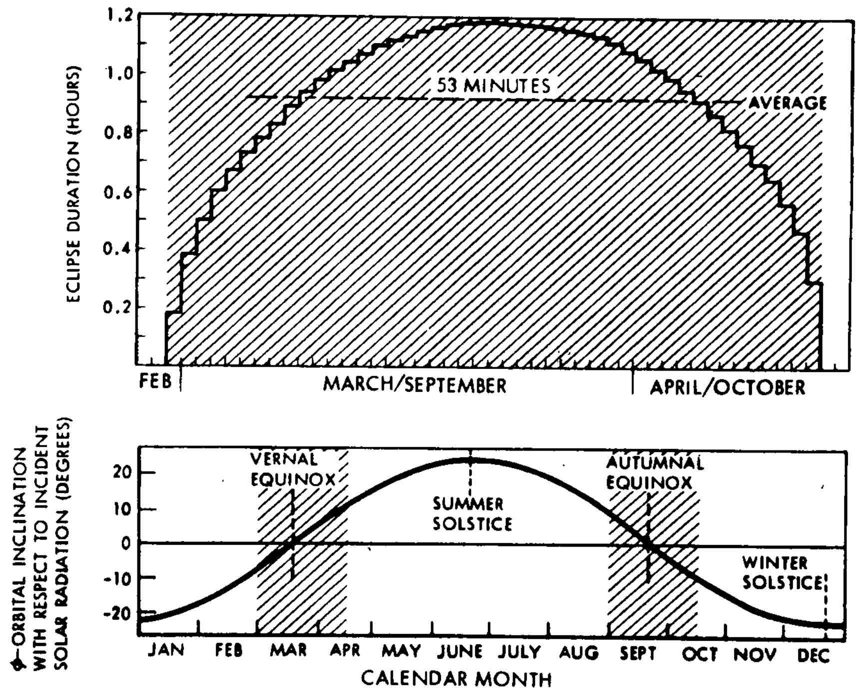

Figure 3.3-27. Eclipse Periods

Figure 3.3-28 shows the duty cycle variation capability of a coated surface versus a louvered surface. Given the maximum and minimum allowable operating temperatures with the surface exposed to the sun and shade, respectively, the allowable turndown ratios (maximum divided by minimum heat load) can be determined. Modules with turndown ratios of approximately 2 to 1 require only a surface coated by an optical surface reflector (OSR). Higher turndown ratios in the range of 10 to 15 require louvered surfaces. Still higher turndown ratios would require the installation of heaters to assure that the subassembly is not exposed to temperatures outside of allowable limits.

The rejection area required for each module is dependent on the average temperature of the surface and whether it is louvered or simply coated. Figure 3.3-29 shows the rejection capability of a square foot of each type of surface facing the sun or deep space. Low-powered modules need only one surface for adequate rejection. The full solar incidence curves should be used to size the rejection area. If an additional surface is necessary for heat rejection the data for the surface away from the sun can be utilized. Final adjustment of these sizes will be necessary to accommodate the worst sun angle for combined surfaces. Variation in operating temperature results from the vehicle passing from sunshine into the shadow of the earth. A surface passively controlled will drop approximately 20 F. Louvered surface temperatures will vary as a function of louver opening as well as external environment. Figure 3.3-30 shows the temperature control capability of the recommended louver design.

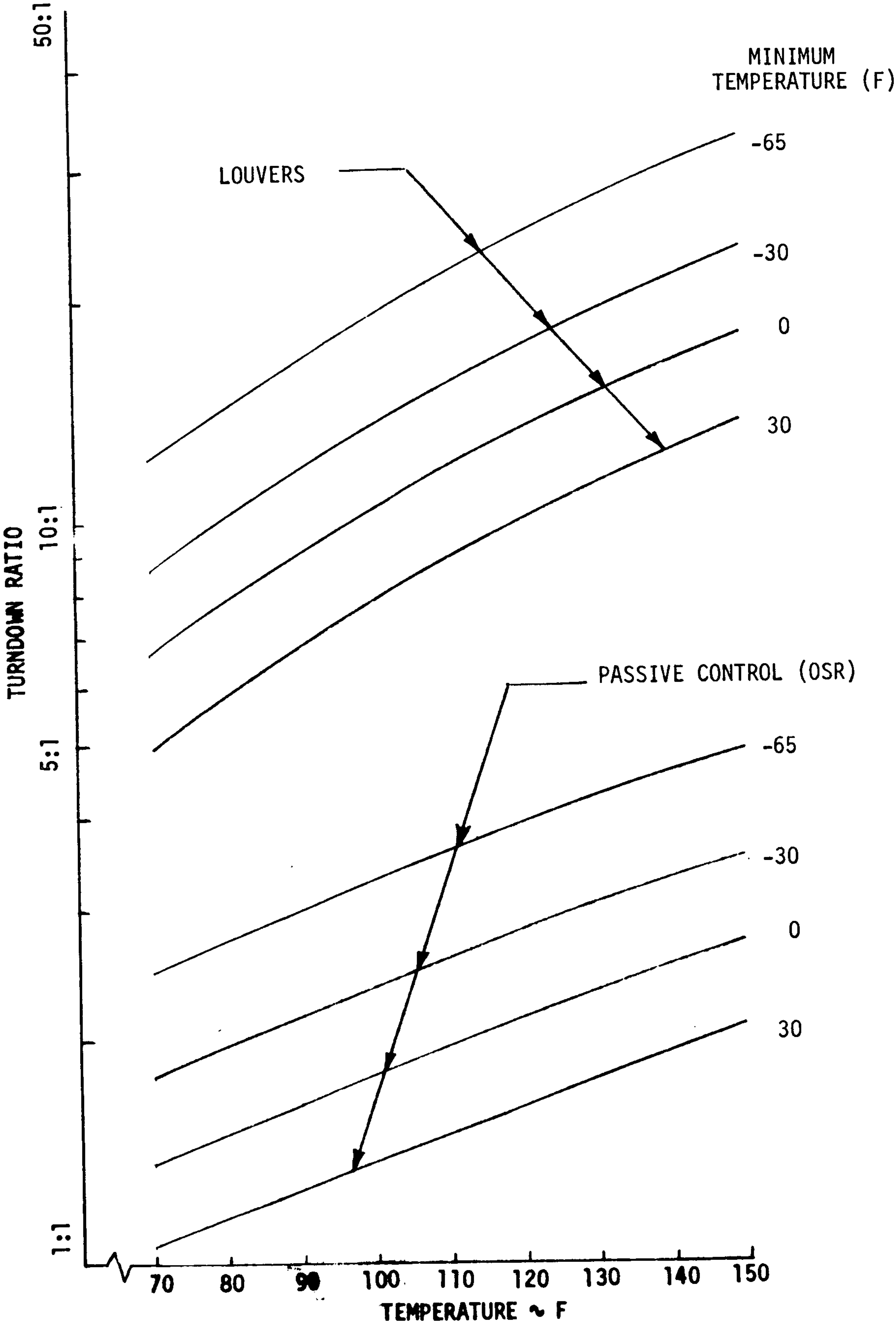

Figure 3.3-28. Power Turndown Ratio versus Maximum Allowable Temperature

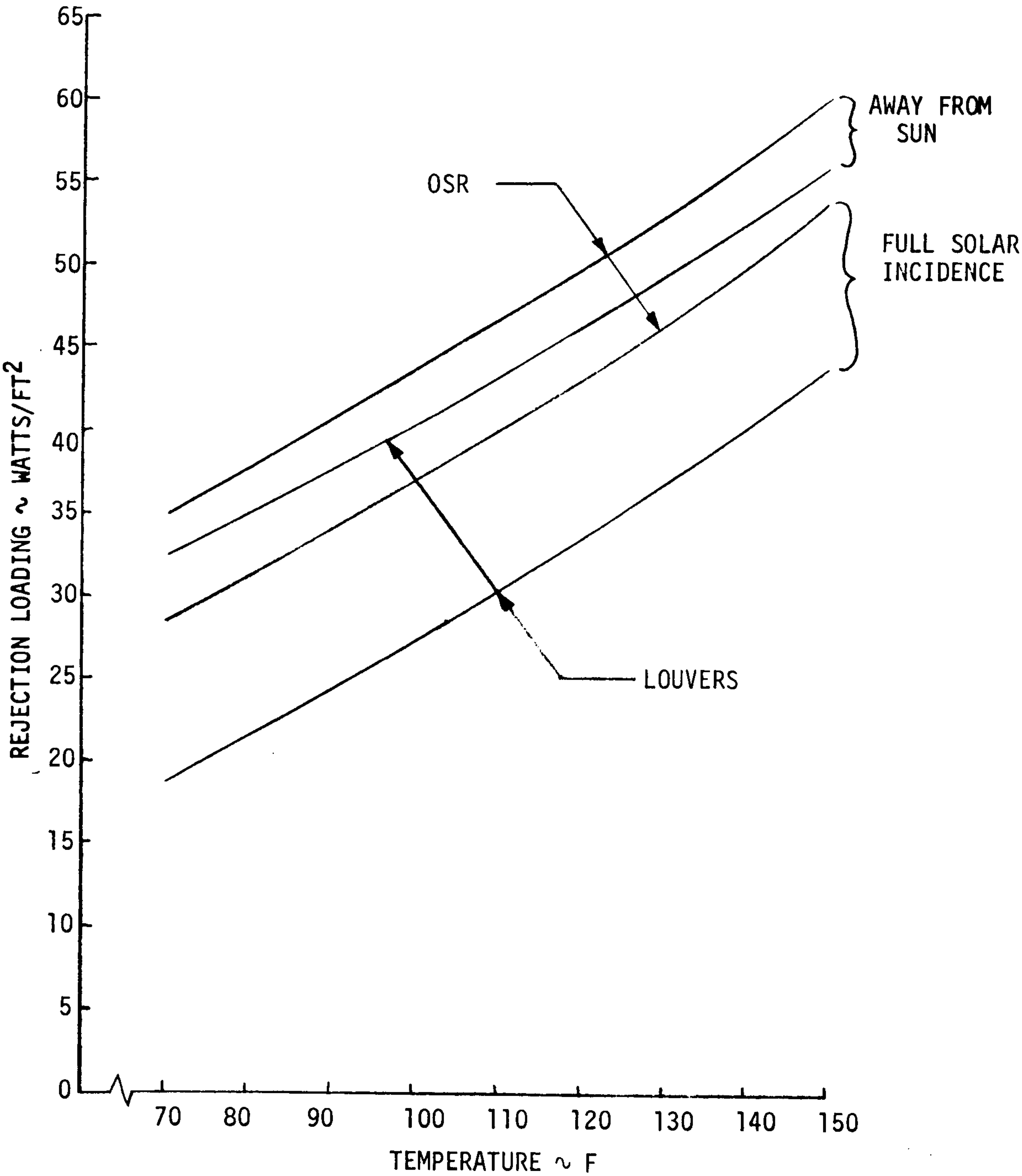

Figure 3.3-29. Radiator Heat Rejection Loading versus Maximum Temperature

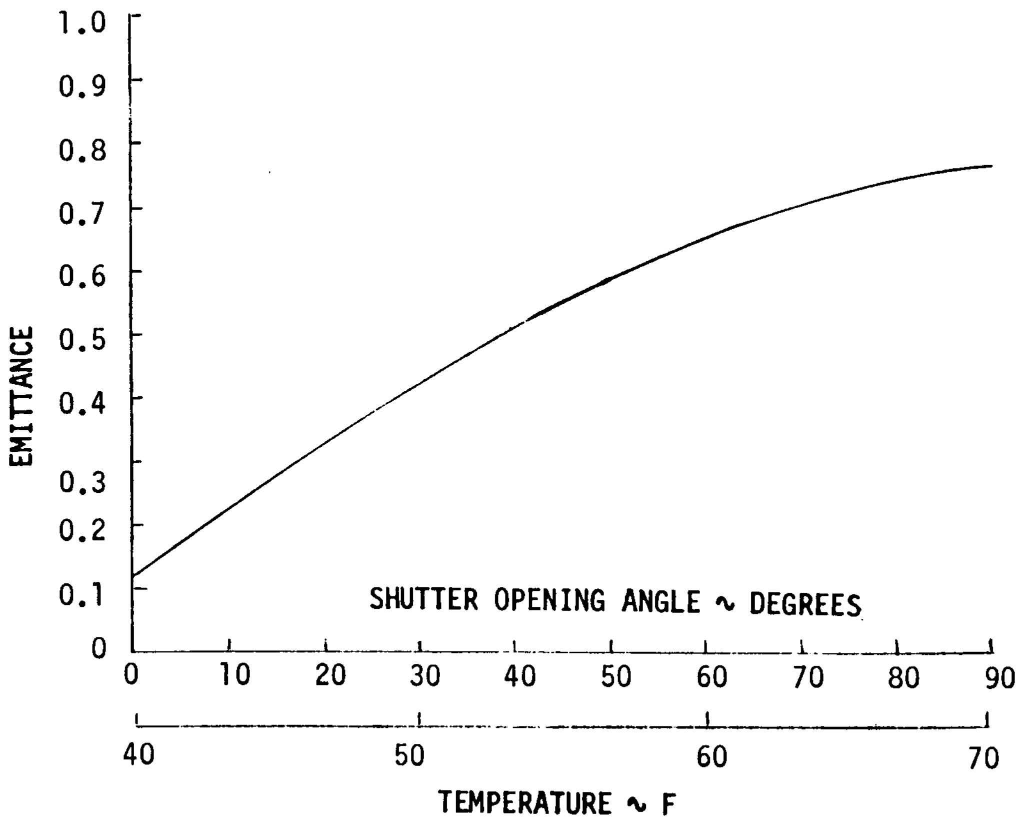

Figure 3.3-30. Louver Operating Characteristics

Thermal Control of Evolutionary Platform Design

The evolutionary design of the geosynchronous platform provides for initial maintenance by auto-remote servicing and later maintenance by men in a shirtsleeve environment. Construction requirements defined by shirtsleeve maintenance drive the design of the TCS. The pressure hull eliminates direct interaction of heat dissipating equipment and the space environment. Leakage limitations dictate few penetrations resulting in a header approach which collects the dissipated energy and transports it to an external surface radiator. The two key design issues to be addressed are (1) what is the optimum thermal control approach for this configuration, and (2) what is the best approach to thermal interchange between a replaceable module and the heat transport assembly.

Thermal Control Subsystem

The infrequency of maintenance visits dictate adoption of a highly reliable TCS concept. Passive type approaches similar to those utilized on current satellites appear to have limited application since a pressure hull isolates the equipment from the space environment. It is doubtful if pumped fluid approaches that have been utilized on all manned spacecraft can achieve the desired reliability. Concepts based on heat pipe energy transport provide the needed reliability while performing similarly to a pumped fluid concept in that energy can be collected and transported to a single penetration of the pressure hull.

Applicable heat pipe concepts differ only in the method of heat rejection control. Feeder heat pipes pick up the thermal energy from each subsystem module and transfers it to a header heat pipe. The header heat pipe transports the energy to an external surface radiator. Heat pipes distribute the energy over the radiator surface to be rejected to space. Radiator control is accomplished between the radiator heat pipes and the header heat pipe. Diode and/or variable conductance heat pipes provide heat rejection control.

Figure 3.3-31 shows the selected TCS concept for the evolutionary platform configuration. Energy is transported from the header heat pipe to a variable conductance heat pipe which, in turn, transports the energy to the radiator heat pipes. The variable conductance heat pipe can accommodate turndown ratios up to 50 to 1 depending on the number of radiator tubes attached. When the internal heat load is at a maximum the vapor heat transport medium expands, compressing the gas in the reservoir. Vapor condenses along full width of the radiator panel. As the heat load diminishes the temperature of the heat transport vapor drops. The vapor contracts and the gas/vapor interface moves toward the header heat pipe interface end. As a result, radiator heat pipes are shut off, reducing the effective radiator rejection area. The gas reservoir is attached to the pressure bulkhead to maintain the gas at close to a constant temperature.

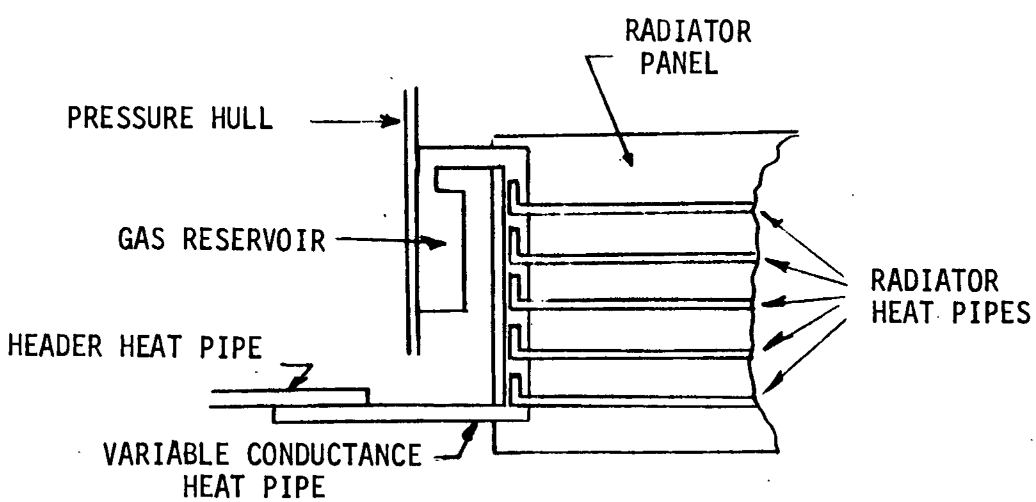

Figure 3.3-31. Evolutionary Platform TCS Concept

Figure 3.3-32 shows the relative performance of alternative heat pipe working fluids. Ammonia was selected for the internal heat pipes since it displays the best characteristics between 0 F and 100 F and does not present the freezing problems common to water. Radiator heat pipes will use Freon 22 as a working fluid. Freon 22 has a freezing point of -256 F which provides adequate margin for shutdown tubes exposed to a deep space environment. This fluid operates at a pressure, 80 to 200 psia, assuming a radiator operating temperature between 40 F and 100 F which does not impose significant structural problems.

A reflective coating is recommended for the radiator surface. Possible candidates are aluminized or silvered Teflon or back-silvered microquartz. All have a solar equilibrium temperature below 40 F, eliminating concern for control of radiator panels which are net energy absorbers and eliminating a need for diode heat pipes. Selection of the coating is dependent on the available radiator area and the generated heat load. Platforms with ample radiator area can utilize aluminized or silvered Teflon coatings (α/ϵ = 0.2 and 0.13, respectively) which are cheaper and lighter in weight. Back-surfaced microquartz has demonstrated performance of an α/ϵ = 0.07 without appreciable degradation.

Module Thermal Interchange

The interface between the subsystem module and the thermal control subsystem is a critical design issue for two reasons: (1) intimate contact is required to reduce the temperature between the electronics and the heat sinks, and (2) the boxes must be easily removed in a zero gravity environment which makes screwed-down pressure type contact surfaces less desirable.

Figure 3.3-33 illustrates three alternative module/TCS interface configurations. Configuration (a) relies on radiant interchange with a shroud which surrounds the module. Since the heat is transferred by radiation, the sides of the module can be used to reject heat while the back can be used for other subsystem interfaces such as electrical connectors. Configuration (b) utilizes an approach similar to that adopted for the Apollo coldplates. To ensure good heat transfer an interface material such as conducting grease, soft metal wools, and coiled copper have been examined. The grease could possibly evaporate in the prolonged vacuum environment. However, since the grease has demonstrated the best performance a test program is required to determine the effects of prolonged exposure to vacuum. Configuration (c) utilizes a heat pipe probe which would be effective in reaching a local internal heat generator. Proper choice of heat pipe working fluids can cause improved contact in a vacuum with a loose fit under atmospheric pressure. The internal pressure in a rectangular cross-section heat pipe forces the flat surfaces against the contacting surfaces of the module when the internal volume of the spacecraft is evacuated.

Figure 3.3-34 shows the relative performance of radiation interchange and contact interchange technique. Conduction transfer aided by a conductive grease is two orders of magnitude better than radiant transfer and approximately five times better than dry contact. Dry contact is the preferable approach if the transfer area is adequate for the heat generated. If the area is inadequate

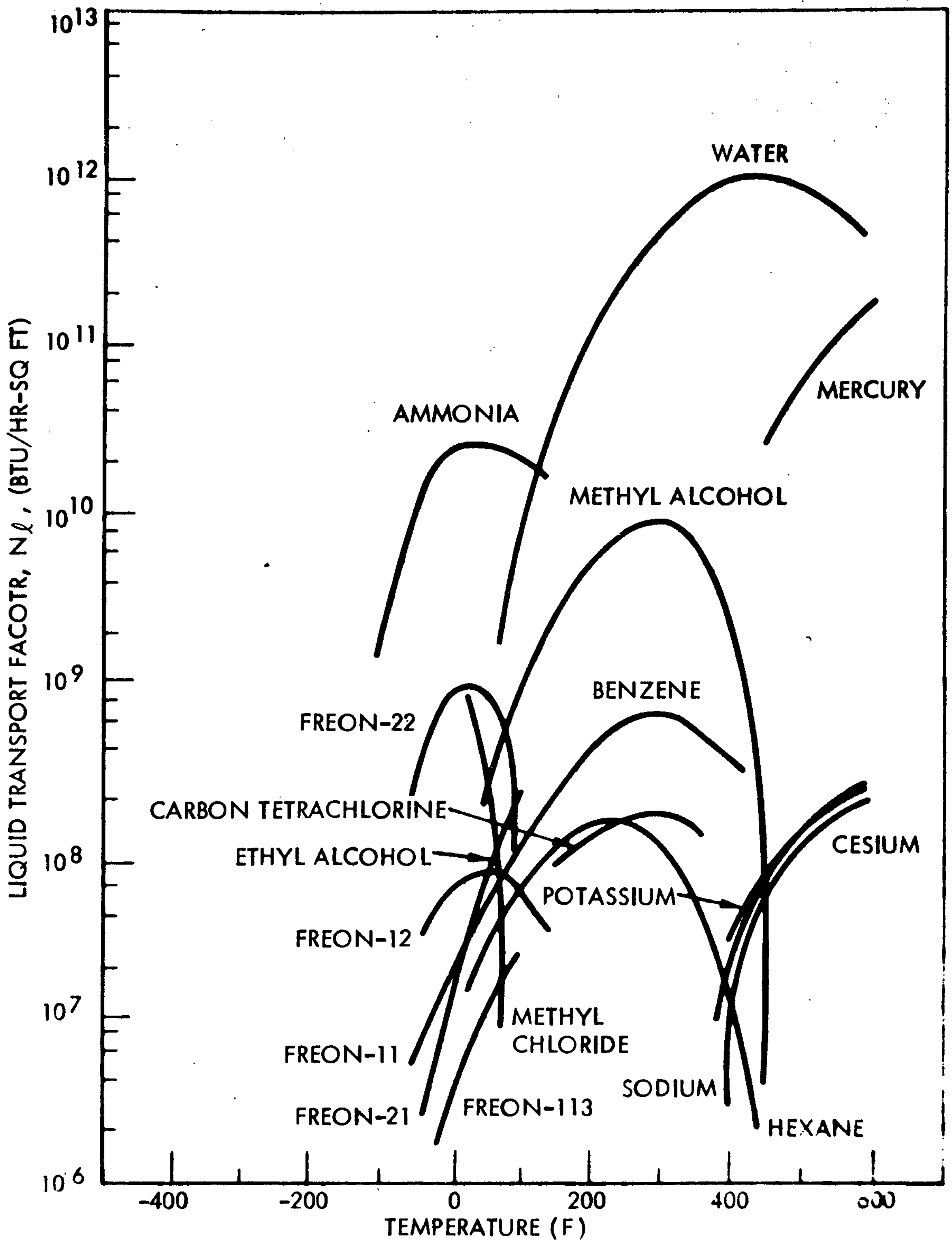

Figure 3.3-32. Liquid Transport Factor for Various Heat Pipe Working Fluids

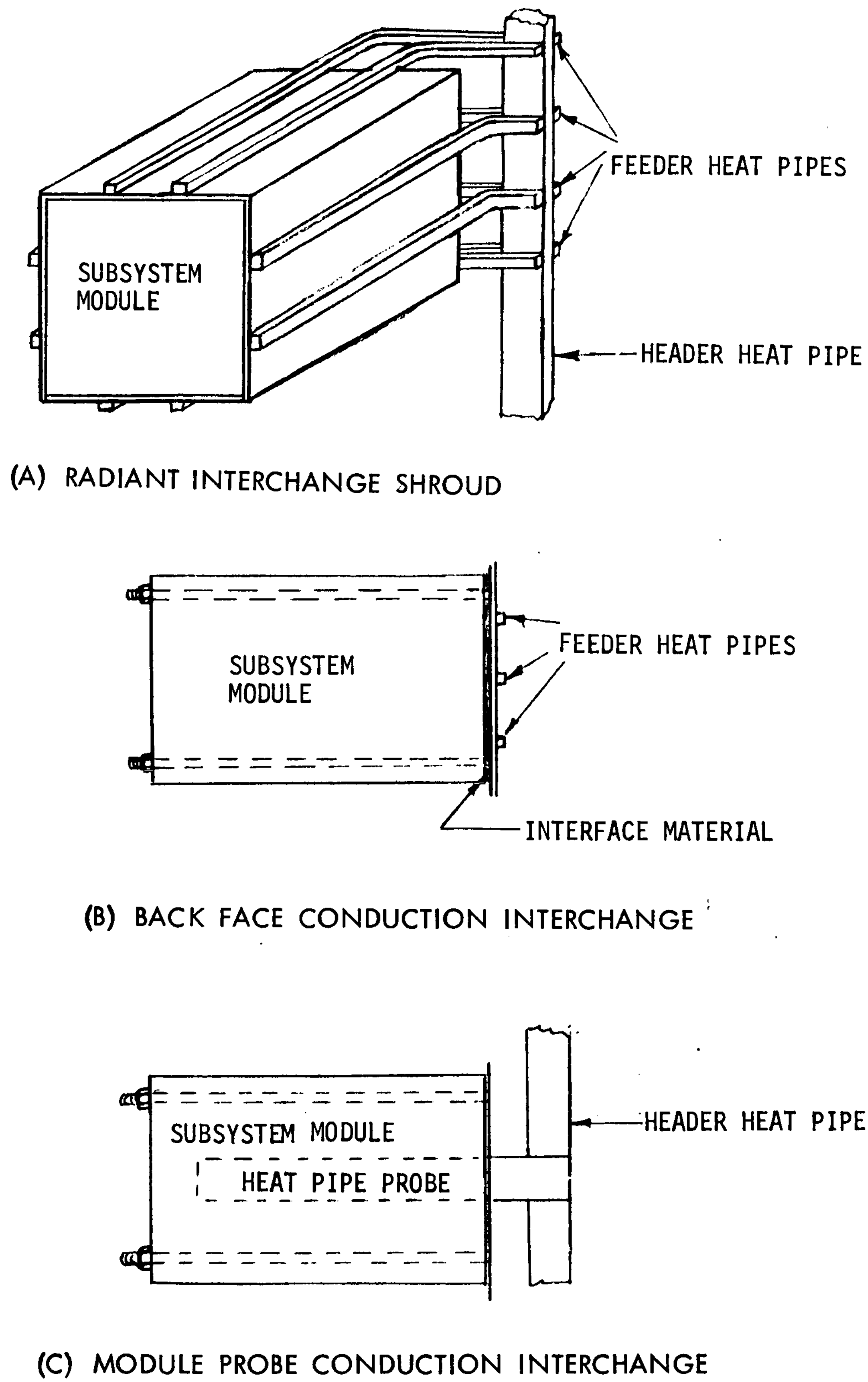

Figure 3.3-33. Replaceable Module Heat Absorption Approaches

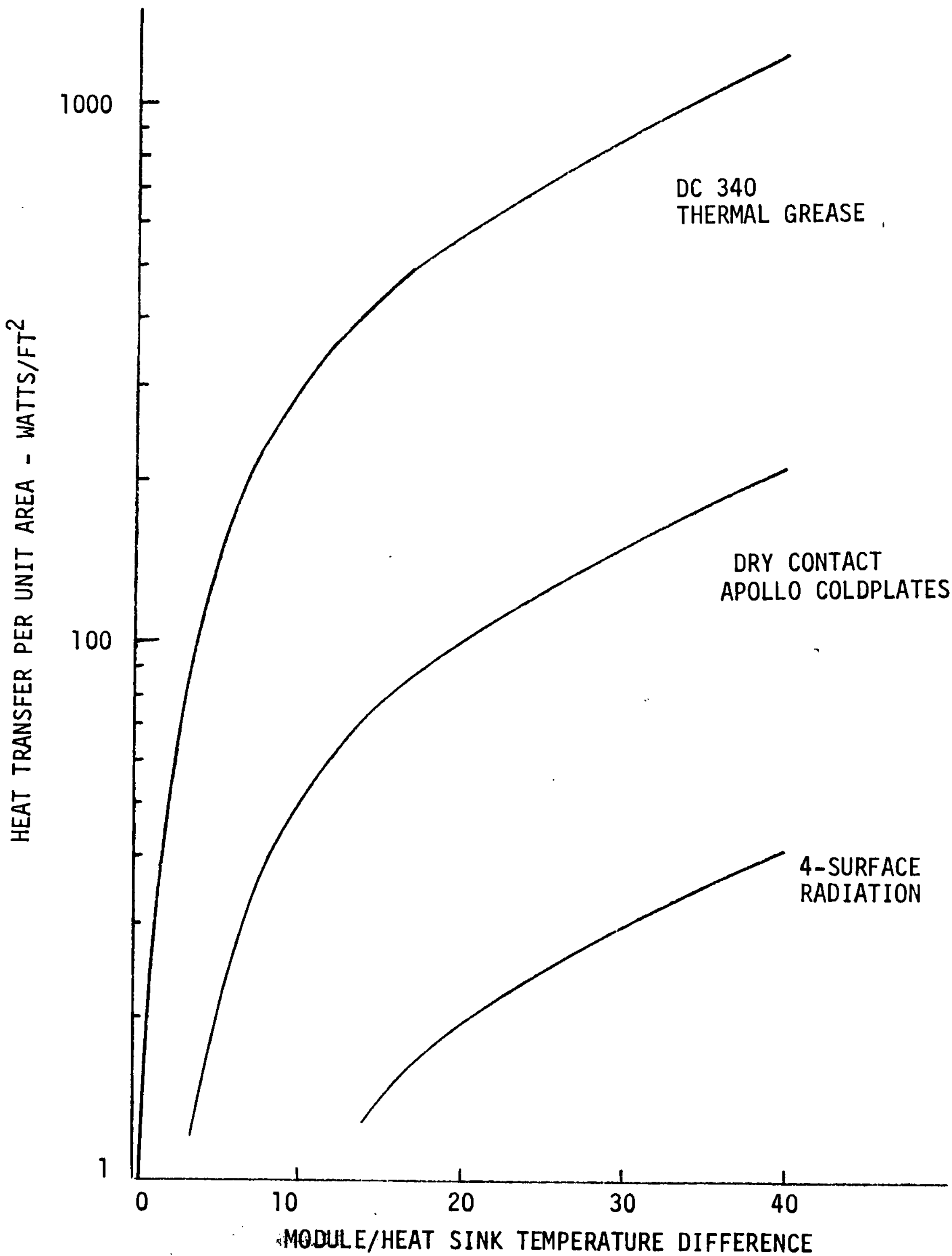

Figure 3.3-34. Heat Transfer Rate Across Module/Heat Sink Interface

conductive grease should be applied. When a module is replaced a new layer of grease would be required. The grease can be encased in mylar and attached to the rear of a module to ensure good contact for automated maintenance activities.

Weight of TCS Concepts

Weight of the TCS differs significantly for the alternate concepts that have been described. Table 3.3-20 presents a comparison of the concept weights as a function of required radiator area. The required substructure weights are included. The simple passive concepts are the lightest weight because they essentially consist of only a coating on the substructure or modules. Two louver concepts are identified to reflect the difference between louvers that directly expose the interior modules to deep space and those that regulate the exposure of a coated surface interior to deep space.

Table 3.3-20. TCS Concept Weights

Concept	Weight ($lb\text{-}ft^2$)
Passive (white paint)	0.57
Passive (OSR)	0.60
Louvers	0.85
Louvers with covered substructure	1.45
Heat pipes	0.94
Heaters	0.83
Thermostats	0.08 each

PREFERRED TCS CONCEPT

The preferred TCS concept is to use heat pipes to transport heat to an externally mounted space radiator. Sizing of the selected concept is dependent upon the operational profile and specific design of the platform equipment.

Common Support Module

The operational timeline of the common support module is assumed to be continuous. Therefore, the TCS for this major assembly can be accurately defined. It consists of six heat pipe coldplates with only dry contact between replaceable modules and the coldplate except for the power distribution module. Interface grease is used between this module and the coldplate. Two header heat pipes transfer the internally generated heat to variable conductance heat pipes which in turn transfer the energy to circumferential radiator panels. The required radiator area is 22 square feet. It is assumed that a coating of 0.005 Teflon coated with aluminum is used, and the radiator operating temperature is 35 F. Table 3.3-21 presents the TCS weight breakdown for the common support module ring.

Table 3.3-21. Common Support Module TCS Weights

Component	Weight (lb)
Coldplates	4.1
Interface material	0.5
Header heat pipes	5.3
VCHP	3.1
Radiator	21.0
Mounts and supports	5.0
Total	39.0

Mission Equipment Modules

All data relay platform concepts are assumed to operate on a continuous basis. The complement of equipment varies between platforms but the basic units that make up the equipment inventory are:

- L-band transponder
- C-band transponder
- K_{LO}-band transponder
- K_{HI}-band transponder

All four replaceable modules require sufficient heat dissipation to warrant inclusion of the interface grease for the thermal interface. The weight summaries for each platform reflect the appropriate number of coldplates and radiators required by the specific complement of mission equipment.

The mission equipment for the observational platforms is considered representative of the type of equipment that will be used during the 1980 time frame. Therefore, only a scaling of the TPS requirements for the common support module was used to derive TPS requirements for the observational platforms. Table 3.3-22 summarizes the TPS equipment for each mission equipment ring of these platforms.

Table 3.3-22. Observation Platform TCS Concepts

Platform	Average Power (watts)	TCS Concept		
		Number of Thermo Grease Interfaces	Radiator Area (sq ft)	Weight (lb)
Earth Observation				
Direct-view sensor ring	524	4	36	65
Telescope and instrument ring	1054	6	72	130
Stellar/X-Ray Astronomy				
Direct-view sensor ring	600	2	22	39
Telescope and instrument ring	184	0	15	20
Solar Astronomy				
Direct-view sensor ring	329	0	22	39
Telescope and instrument ring	670		44	80
Plasma Physics				
Mission equipment ring	650	3	44	80
High-Energy Physics				
Absorption counter ring	555	0	44	80
Sensor ring	25	0	*	*

*No unique requirements; therefore, common support module TCS concept can be used.

ENVIRONMENTAL CONTROL SUBSYSTEM

Incorporation of environmental control subsystem (ECS) equipment in the geosynchronous platform was evaluated to facilitate shirtsleeve servicing of platforms. Because of the potentially long duration (years) between servicing visits and the ECS requirements of the manned tug itself, it was not practical to include life support provisions on the platform. Therefore, all ECS equipment required for manned servicing of platforms is considered part of the crew module of the tug. The delta tug ECS requirements for platform servicing are defined below.

Atmospheric Composition

Humidity control, CO_2 control, module total pressure, and oxygen partial pressure control requirements are defined by the size of the maintenance crew and their need for an earth-type environment. None of these change when they move from the tug to the platform and therefore no additional equipment for these functions is required.

Temperature Control

Atmosphere temperature control between 65 F and 75 F is accomplished by a heat exchanger and fans located in the tug. The capacity of the heat exchanger must be increased 1700 Btu/hour to accommodate the additional equipment heat load picked up by the air in the geosynchronous platform. The fan capacity must be increased by 150 cfm to provide adequate ventilation to the platform. Air is then ducted into the platform volume from the tug. Louvers on the ducts of the tug can be adjusted to increase flow into the platform if necessary. Air is returned to the tug through the connecting hatch. Portable fans can be placed at locations on the platform where the maintenance crew is working to meet local velocity requirements of 20 to 40 feet per minute.

Pressurization

Prior to entry into the platform service area the volume must be pressurized. The quantity of air required is dependent upon the volume of the pressurized compartment and the number of repressurizations required. This quantity varies with each platform. Likewise, the number of high-pressure tanks required to store this gas is dependent upon the number of platforms to be visited in any one mission. The optimum storage pressure is 3000 psia and the weight is generally assessed at one pound/pound of air stored. An atmosphere pumpdown system on board the platform was rejected because high-pressure gas cannot be stored for the long durations required without depletion from leakage.

Depressurization

Subsequent to maintenance activities the platform should be depressurized prior to platform disengagement. This action prevents pressure vessel leakage from affecting attitude control of the platform. The vent valve required to dump the platform pressure will be remotely activated from the tug. The valve is sized to limit the depressurization time to approximately one-half hour.

Fire Detection

Fire detection is accomplished by a condensate nuclei counter in each module. A sample of air from the duct system is passed through the detector where the number of particles in the air sample is determined. When the particle count reaches a predetermined level, an alarm signal is given of a pending fire hazard. The basic principle behind the particle counter approach is that all materials emit large amounts of particles when materials are approaching ignition temperatures.

Hardware Complement

The required ECS tug hardware for support of shirtsleeve servicing of platforms is summarized in Table 3.3-2_. The size and quantity of atmospheric storage tanks are dependent upon the servicing duration at a platform and the number of platforms visited per tug mission. Therefore, it is recommended that the tanks be incorporated in the replacement module storage section of the manned servicing configuration rather than be an integral part of the tug crew module.

Table 3.3-23. ECS Hardware Characteristics

Component	Quantity	Unit Weight (lb)	Size (in.)		Power (watts)
			Length	Diameter	
Duct	1	6	84	8	
Duct coupling	1	10	9	5	
Ventilation fan	2	2	4	6	15
Fire detector	1	6	6	6	10
Platform vent valve	2	3	2	4	
Pressure tanks		1 lb/lb air			
Consumable					
Air		7.5 lb/100 ft^3			

SUMMARY

The selected support system concepts, the attendant weight and power of the replaceable modules, and the common support module structural weights are summarized in Table 3.3-24. The weight of the integrated common support module will vary depending upon the mission equipment rings with which it is combined. The primary variables are (1) solar array booms, (2) solar array and battery pack, and (3) the docking mechanism. The booms are required on some platform to extend the solar arrays beyond the mission equipment sensor/antenna complex. Array and battery pack size are dependent upon the power requirements of the mission equipment. In some configurations docking may occur with mission equipment rings rather than the common support module.

Table 3.3-24. Integrated Common Support Module Summary

Item	Weight (lb)		Power Required (watts)	
	Subtotal	Total	Subtotal	Total
Structure		500		--
Primary	100			
Secondary	300			
Docking Mechanisms	100			
Electrical Power		1251		70
Solar Array Assemblies (2 KW)	240		10	
Power Conditioner	144		20	
Battery Pack	392		40	
Solar Array Booms	300		--	
Cabling	175			
Data Handling		70		28
Electronics	60		28	
Antenna	10		--	
Attitude Stabilization & Control		429		120
Reaction Wheels	75		30	
CCD Star Trackers	30		20	
Flight Control Electronics	44		50	
RCS Quads (4)	280		20	
Thermal Protection		39		
Radiators	26			
Coldplates, Heat Pipes, etc.	13			
Totals		2289		218

3.4 ON-ORBIT SERVICING CONCEPT

Based upon the design criteria established for on-orbit servicing (Volume IV, Part 1, Section 5.4) and the preferred common support module structural configuration developed in Section 3.1 of this volume, the conceptual designs for the servicing systems are derived in this section. Although standardized packaging is the preferred mode, an option compatible to the basic concept is also presented. The ancillary equipment associated with the three servicing modes and the sequence of operations are described.

STANDARDIZED PACKAGING

The requirement for standard subsystem modules is generated by the use of an auto-remote servicing system. Different sized boxes or different types of attachment features would unduly complicate the mechanization of a servicing system. Either adapters, multiple attach points, or additional storage volume would be required in the servicing unit.

Size requirements for each subsystem were investigated and the module with the largest volume requirement was found to be the RCS (the large propellant tank). A further constraint on size is packaging twelve modules within the 12-foot diameter common support module structure. These requirements led to the selection of 24 x 20 x 24-inch standard module. The volume provided by a module of this size was found to be adequate for most mission sensors and electronic assemblies. In fact, some of the volumes required by sensors fell into the range of one-third of a standard module. Thus, it may be advisable to also develop two or even three smaller modules that fit within the standard module envelope. The possibility of the division of a module into two or three separate modules suggested that a restriction be placed on the lock mechanism to two places per module. Although four locking mechanisms probably would provide better structural efficiency, two were determined to be acceptable. As shown in Figure 3.4-1, the standard module may be divided into a variety of shapes providing the lock mechanism retains a standard dimension. This allows flexibility in design and placement of mission equipment sensors and electronics.

SERVICING CONCEPTS

In order to establish the feasibility of servicing and refurbishment of a platform with a circular ring type body as servicing evolves from auto-remote to manned systems, it was necessary to develop some conceptual approaches for the operations. Figure 3.4-2 shows two approaches for servicing a platform using unmanned techniques. Concept A assumed that equipment modules inside the body are the only ones to be replaced while Concept B required replacement of exterior assemblies in addition to the interior modules. It is shown that the servicing system approach for A and B concepts does not change; only the length of the manipulator arm increases in size. One other important aspect of design is shown for Concept B. If exterior platform assemblies are to be exchanged and these assemblies cannot be attached to the outside of the

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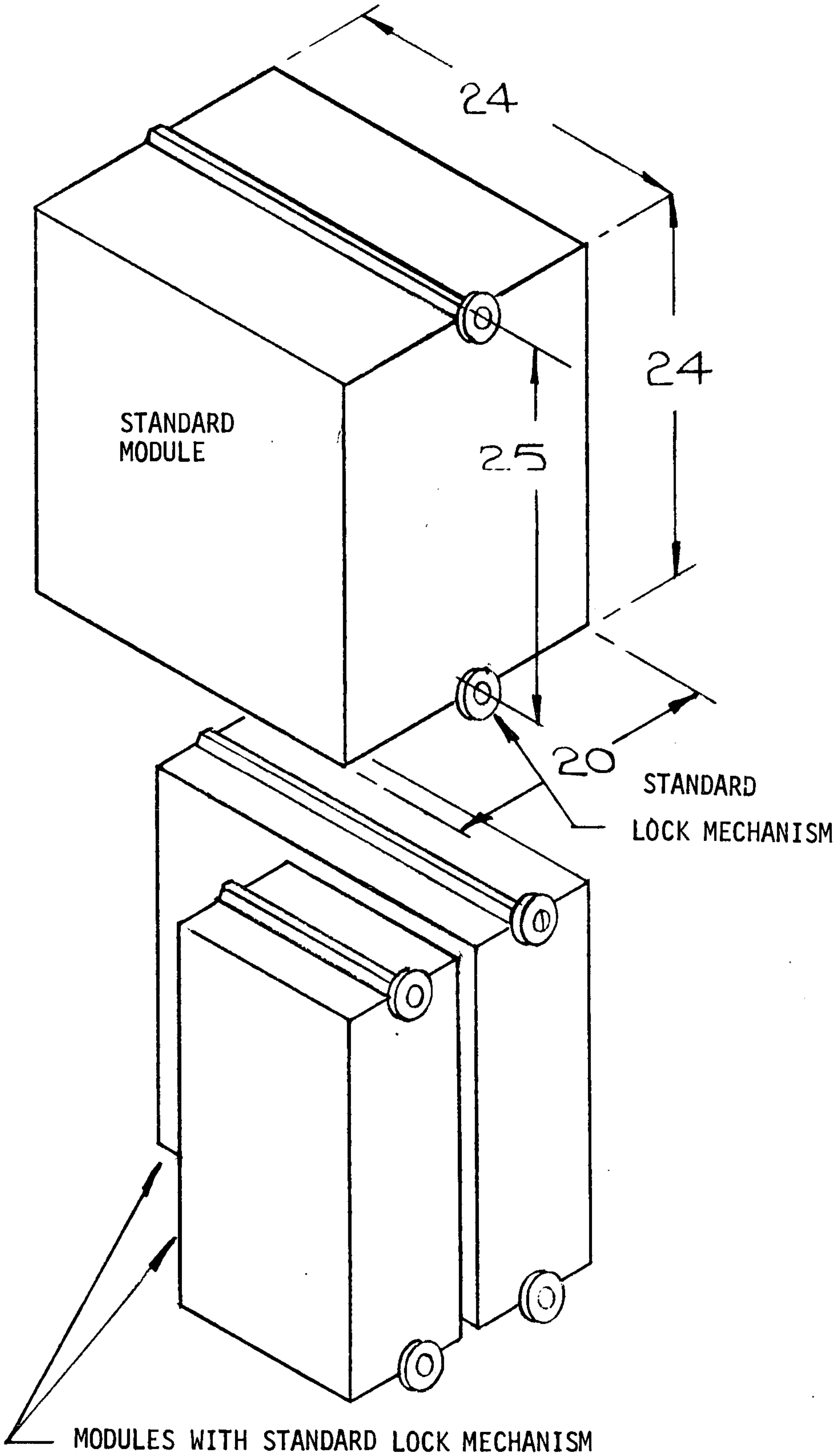

Figure 3.4-1. Standard Modules

C

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servicing system because of volumetric restrictions in the space shuttle cargo bay, these assemblies must be transferred from a temporary hold position in the payload bay to the outside of the servicing system after the tug has been deployed. This must be performed with the shuttle attached manipulator.

Concept C shows evolution to a manned servicing system where dual tugs are required to achieve synchronous orbit. Note that exterior assemblies which cannot be stowed on the exterior of the crew module must also be stowed on a temporary hold fixture in the shuttle bay.

Auto-Remote Servicing Concept

The selected conceptual approach for remote servicing is shown on Figure 3.4-3. The system features a variable number of tiers for replacement module stowage (two tiers are shown), a docking device, and a manipulator to perform module exchange. A second articulating arm is also included to provide TV and light for ground control observation and direction of the operation. In addition to the module exchange task, the system has the capability of replacing solar panels and antennas.

The servicing system is launched within the shuttle bay and attached to the tug through a short structural adapter. The manipulator and TV arm are stowed forward of the module rings. They are attached to the shuttle structure through a saddle fitting. An additional attachment with pyrotechnic release is provided to support the manipulator terminal device from the module rings. During docking to an on-orbit platform,the manipulator and TV arm are swung away to provide adequate clearance for the docking maneuver.

After docking to the platform, the docking ring along with the platform is rotated 90 degrees and locked in place. This enables the manipulator and TV arm to have access to both the platform and service system interior and exterior to perform the replacement function.

The manipulator is designed to use shuttle-attached manipulator hardware with attendant cost saving. Because the manipulator is considerably shorter than its shuttle counterpart, it will be considerably more rigid during operation. This is required because of the more delicate nature and accuracy required to replace modules.

Replacement operations consist of unlocking the module from the platform structure and withdrawing it in a radial direction. The module is then transported to the service system ring and placed in an empty compartment. A new module is extracted from the ring and placed in the platform. The sequence is simply repeated for all the affected modules. The modules are provided with rails which match rails on both the service system storage bay and the platform structure. Coarse operations such as transporting modules and initial alignment are controlled from the ground via a television link. The television camera is mounted to the tip of a separate arm together with a pair of floodlights in such a manner that the assembly can be gimbaled.

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During module insertion any binding in the rails due to misalignment is sensed as a load in the manipulator rotary joints. The manipulator control system incorporates a force feedback system to allow the operator to "feel" the forces and moments.

The terminal device of the manipulator is required to provide the forces to operate the module attachment mechanism and at the same time grasp the module for extraction and transportation to the servicing system. The conceptual design for the terminal device mechanism is presented in Section 3.5. The design features an outer sleeve which moves axially to deflect a number of fingers over the module attachment fitting. Axial forces for unlocking or locking the module mechanism (in the order of 50 pounds) and for compressing the preload springs (approximately 1000 pounds) can then be applied simultaneously to both module mechanisms.

Table 3.4-1 presents a weight summary of the ancillary equipment required for the auto-remote servicing modes.

Table 3.4-1. Auto-Remote Servicing Unit Weight Summary

Assembly	1 Tier (lb)	2 Tier (lb)
Docking mechanism	100	100
Manipulators and TV arm	400	450
Structure	400	800
Adapter	100	150
Unit subsystems (TV electronics, TPS, docking aids)	150	150
Total	1150	1650

Shirtsleeve Servicing

Servicing of interior platform subsystems may be accomplished in a shirtsleeve environment when the platform is designed to be pressurized; however, servicing or replacement of exterior platform appendages may require EVA and/or manipulator operations. Figure 3.4-4 conceptually defines a manned system having both EVA and shirtsleeve servicing capability simultaneously. The crew module consists of a pressurized cylindrical structure approximately 160 inches long and 144 inches diameter. The upper forward position is devoted to the pilot's compartment and associated equipment. The volume under the floor provides a passageway between the replacement module storage area and the platform to be serviced. This section is also used for EVA egress/ingress operations and crew module subsystems installation.

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The tool used to remove a subsystem module from the platform is a hand-held version of the terminal device described for the auto-remote servicing system. The rationale behind a special tool is that (1) preload forces are difficult to attain by hand, and (2) use of the tool provides evolution from unmanned to manned servicing.

The launch configuration illustrated shows the man-rated tug/crew module assembly in the shuttle payload bay with the docking module installed. Sufficient clearance has been allowed for pivoting the payload out of the bay prior to attachment to the docking module for crew transfer.

Table 3.4-2 presents a weight summary of the crew module concept.

Table 3.4-2. Crew Service Module Weight Summary

Assembly	1 Tier (lb)	2 Tier (lb)	3 Tier (lb)
Docking mechanism	100	100	100
Primary structure	800	900	1000
Secondary structure	1450	1750	2050
Subsystems	3000	3000	3000
Adapter	250	300	350
Total	5600	6050	6500

Pressure Suited Servicing

The requirements placed on the platform and servicing system for pressure suited operations are basically the same as that for shirtsleeve operation. The crew compartment should be pressurized during ascent to synchronous altitude and during descent back to the shuttle. It could be evacuated during servicing operations at the platforms. This concept eliminates the requirement for an airlock. For purposes of this study the configuration of the crew module for pressure suited servicing is the same as that for shirtsleeve servicing.

3.5 PLATFORM INTERFACES

During the course of the platform design synthesis activity a conscious effort was made to minimize the number of interfaces and simplify those interfaces that were unavoidable. Several of the interfaces merit special attention. They can be grouped into two categories: (1) intra-platform (replaceable module-structure), and (2) inter-platform (platform-logistics system). Key interfaces of these two groups are discussed in this section.

MODULE-TO-PLATFORM STRUCTURE INTERFACES

The four key interfaces between replaceable modules and the platform structure are (1) mechanical, (2) electrical, (3) RF, and (4) thermal. Each of these interfaces are discussed in the following paragraphs.

Mechanical Interface

The concept for module attachment to the platform structure is by means of two tension "bolts" in addition to support provided by guide rails. In order to prevent motion of the module relative to its attachment structure, it is good design practice to preload the bolts up to their expected loading. Table 3.5-1 shows the load factors expected during a shuttle launch. It has been estimated that 1000 pounds preload per bolt will be sufficient to meet these requirements.

Figure 3.5-1 shows the structural aspects of a typical module. It is important that the guide rails and mechanism retain positional accuracy during handling, therefore, machined sides are used. One of the sides forms a door for access to the interior. A seven-inch extension in length is allowable without interfering with the manipulator terminal device.

A simple mechanism is used for module attachment. It consists of a spring loaded bolt or rod with a conventional ball lock feature. A stack of Belleville washer springs provide the preload force. In operation the module is slid along the guide rails by means of the manipulator until the male cone is in contact with the female cone. The locking device is depressed during this operation to allow the balls in the ball lock device to retract so that they offer no resistance to proper mating. The preload rod is then depressed until the balls are to the rear of the female cone. Release of the center locking rod extends the balls and provides the locking feature when the preload rod is released. Depressing the central lock rod results in release of the module. Both latches are operated simultaneously.

Accurate alignment of the rear surface of the module relative to the platform structure is achieved by cone to cone mating under a compression force. The guide rails are designed to be unloaded toward the rear (outboard) of the module. However, at the inboard end the rails are designed to resist a side load. This has the effect of eliminating moments at the outbouard mechanical attachment.

Table 3.5-1. Shuttle Cargo Bay Limit Load Factors*

Condition	X	Y	Z
Liftoff***	2.2 0.9 1.1	+0.2 -0.7 -0.2	0.0 +0.3 +0.8
High-Q Boost	1.9	±0.2	+0.5 -0.2
Booster End Burn	3.0 ± 0.3	±0.2	±0.3
Orbiter End Burn	3.0 ± 0.3	±0.2	±0.4
Space Operations	+0.2 -0.1	±0.1	±0.1
Entry	±0.25	±0.5	-2.5 +1.0
Flyback	±0.25	±0.5	-2.5 +1.0
Landing and Braking	+0.8 -1.0	±0.5	-2.5
Crash**	-9.0 +1.5	±1.5	-4.5 +2.0

*Positive X, Y, Z directions = forward, right, and down. Load factor carries the sign of the externally applied load.

**Crash load factors are ultimate; all others limit. The longitudinal load factor shall be directed in all forward azimuths within 20 degrees of the longitudinal axis.

Crash load factors are for the nominal payload of 40,000 pounds. Payload retention systems for payloads in excess of 40,000 pounds shall possess strength equivalent to a constant nW product.

The specified crash load factors shall act separately.

***These factors are dynamic transient load factors at liftoff.

E

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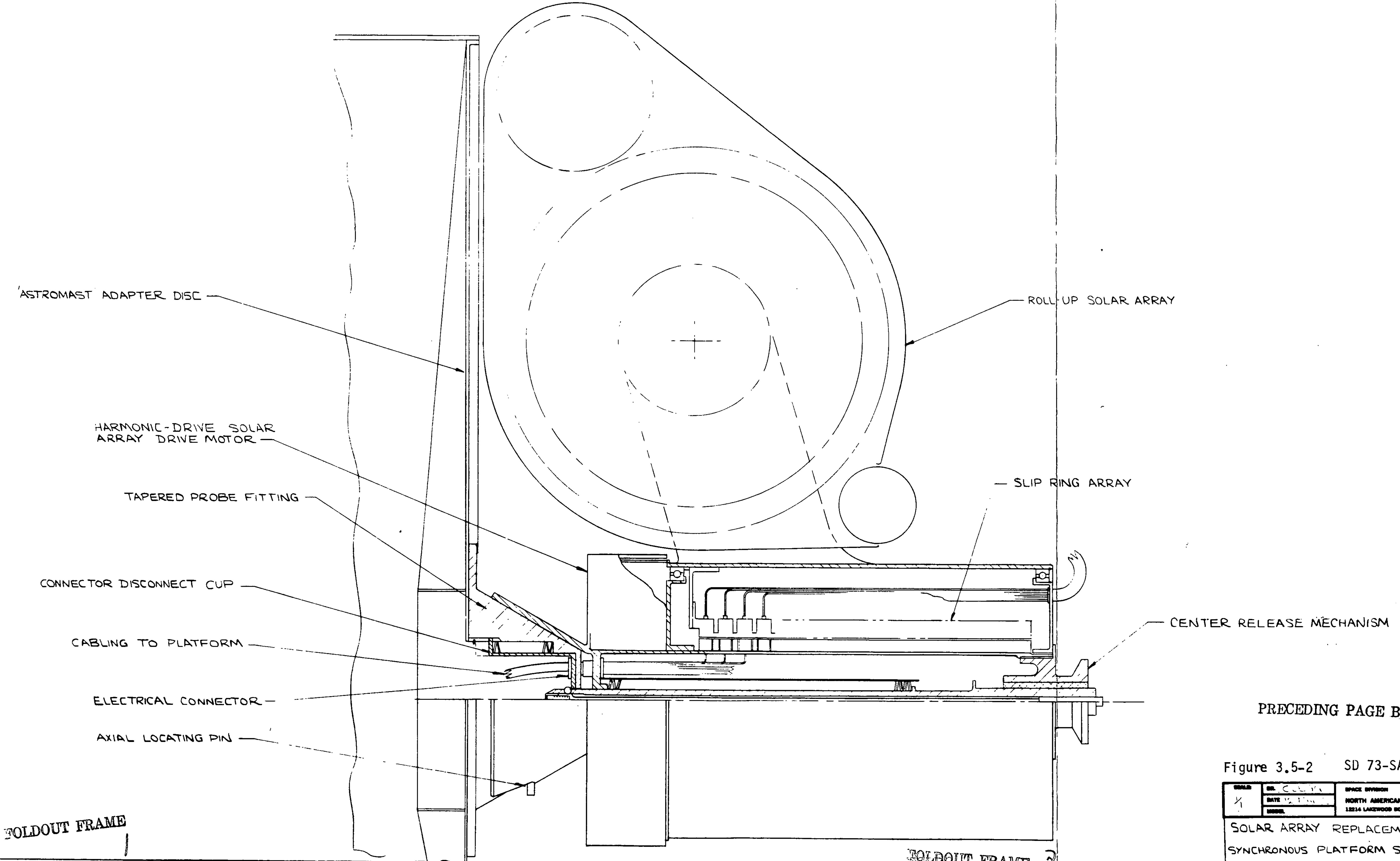
Figure 3.5-2 shows how the module release mechanism can be applied to attach a replaceable solar array using the manipulator and its terminal device. The assembly consists of the roll-up solar array, the array drive motor, and slip ring assembly which transfers power across the rotating joint. The mechanism features a conical probe and cup with axial location provided by a locating pin riding in a tapered slot. The mechanism differs from the module mechanism by the requirement to carry power across the joint. This is accomplished by a number of connectors placed around the central rod. In order to eliminate pin damage in the connectors due to misalignment, the design features connector mating only after correct alignment of the array assembly to the probe fitting. This is accomplished by addition of a "connector disconnect cup" to which one-half of the electrical connectors is attached. The cup is spring loaded with sufficient load to overcome connector mating forces. After correct alignment of the array, the preload rod is depressed until the ball lock is behind the connector cup. Release of the lock rod (to extend the balls) and release of the preload rod results in connector mating. Conversely, depression of the lock rod results in demating of electrical connectors before array removal.

Electrical Interfaces

At the back of each module is an electrical connector. Figure 3.5-1 shows the rear surface of a typical module used in the mission equipment ring of a data relay platform. The interface consists of 16 wave guides, two coaxial RF, and 50 electrical contacts.

Considerable investigation has been performed by the aerospace industry in the field of plug-in subsystems. Experience on military aircraft using plug-in modules with conventional connectors has not been satisfactory. Due to the large forces developed in mating a large number of pins, it is impossible to tell whether a pin is being bent during the process of installation. Generally the pin bends over and contacts an adjacent pin. During subsequent checkout the fault is usually identified as an electronics failure rather than a connector failure. Another aspect of the installation that results in degraded performance but not necessarily a failure, is noise generated in circuits by working of the pins during structural loading conditions.

The advanced plug-in concepts have been used and satisfactory qualified for a few assemblies in the Mariner spacecraft. Subsequently, JPL proceeded with development of plug-in modules for their outer plant spacecraft. Because a data bus was not considered, JPL estimated that 400 interconnect pins would be required on certain electronic modules in order to check out each redundant circuit after module installation. Figure 3.5-3 shows the module design. A total of eight microminiature connectors form the electrical interface. The selected connectors were "Microdot D" qualified for space application. These connectors are shown in Figure 3.5-4 and feature 51 pins on 0.05 centers packaged in a 0.2 x 1.0 metal case. The connectors can be modified to carry two coaxial cables according to the manufacturer. The concept is considered to be developmental because a change in scope of the JPL program prevented completion of development and evaluation.



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FOLDOUT FRAME 3

Figure 3.5-2 SD 73-SA-0036-5 3-121, 3-122

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION
12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

SOLAR ARRAY REPLACEMENT CONCEPT
SYNCHRONOUS PLATFORM SERVICING STUDY

2339-20

FOLDOUT FRAME 1

FOLDOUT FRAME 2

MINIMUM MODULE ENVELOPES

CONNECTORS		DIMENSION	
QUAN	TYPE	X, in.	Y, in.
2	25 P	3.460	0.230
2	37 P	4.060	0.230
2	51 P	3.960	0.275
4	25 P	3.460	0.490

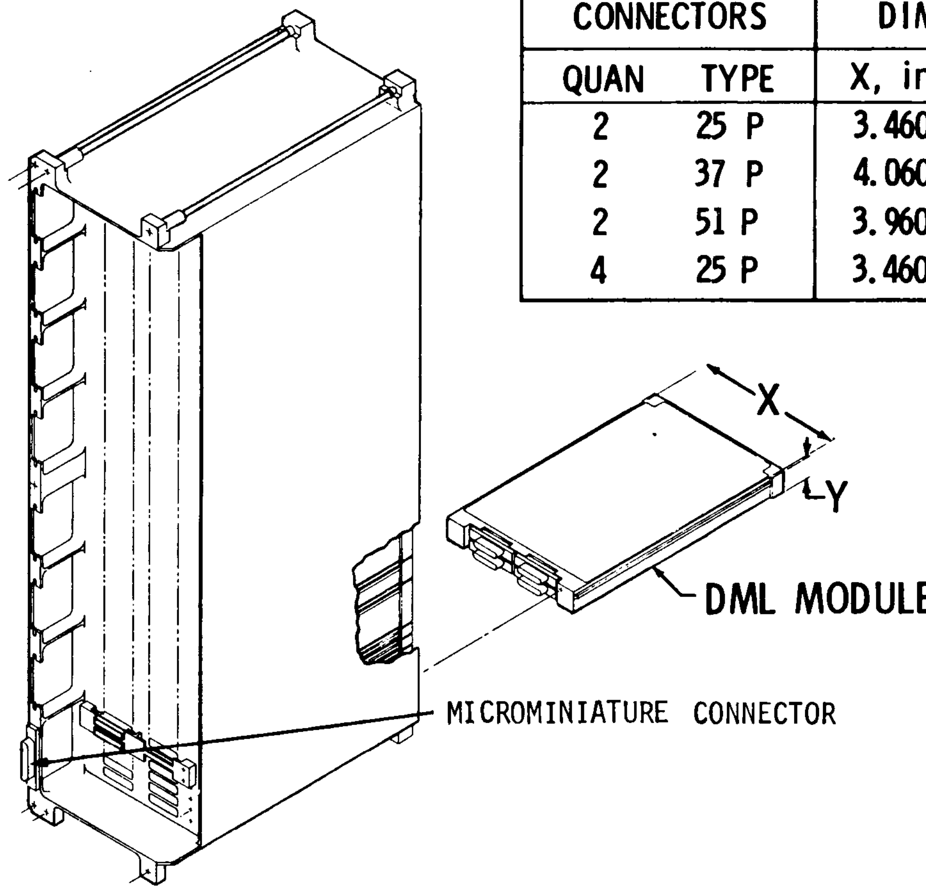

Figure 3.5-3. JPL Plug-In Subsystem Module

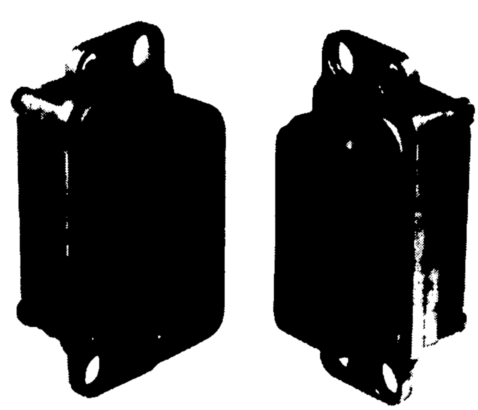

Figure 3.5-4. Microminiature Connector

The selected design for the platform module electrical interconnect consists of two microminiature metal case connectors for 50 electrical pins and two coaxial cables. A floating mount of one-half of the connectors together with lead-in on the metallic cases provides accurate alignment of the pins prior to mating.

RF Interconnections

Figure 3.5-1 also illustrates a waveguide interconnect concept (worst case of 16). The requirements for the interconnect are that each wave guide should match within a few thousandths of an inch after installation. This is achieved by manufacturing the interconnect in one piece by conventional methods, then cutting it in half. Matched master tools are used to locate each half on the platform structure and the module using the conical structural attachments as reference. Tooling is retained so that additional modules can be manufactured to the same tolerances.

Thermal Interface

Section 3.3 presented the cooling concept for replaceable modules. In those cases that the power dissipation of a module exceeds 100 watts, it is recommended that a grease packet encased in mylar be attached to the rear surface of the module. Upon installation, the normal locking forces will crush the bag and provide a thermal short between the module and the coldplate of the platform structure.

PLATFORM TO TRANSPORTATION SYSTEM INTERFACES

Physical interfaces consist of two basic types, structural and docking. These two are interrelated since the lighter weight platforms use the docking ring as a structural attachment to the tug. The heavier platforms are attached to the tug by a conical adapter. This is illustrated by reference to the launch configuration for a data relay platform and an earth resources platform. The heavy platforms are attached to the tug through a conical adapter which provides a more efficient load path than the docking mechanism, while at the same time reduces the length of the payload. The separation mechanism of the adapter consists of six pyrotechnically activated separation devices.

The docking mechanism for an unmanned platform is based on space shuttle hardware. The major difference between the platform docking mechanism and that specified for the shuttle is larger diameter of the platform mechanism. The increase in diameter is required to enable the manipulator and television to operate inside the platform during auto-remote servicing operations. The mechanism consists of four double tapered petals which engage with matching notches in the mating ring. Capture latches on the petals prevent disengagement until the mating halves are drawn together and latched by a series of docking latches. An attenuation distance of 10 inches is provided. As indicated previously, docking a servicing system to a platform requires a television camera and laser radar with target and reflector respectively on the platform for position, range, and range rate determination. This data is transmitted to the ground via the tug communications system.

For manned servicing systems the shuttle docking mechanism is used. One change is to reverse the system so that the active portion of the docking mechanism (attenuators and latches) is on the servicing system. This change allows the inactive half to be on the platform with attendant increase in reliability.

Evolution from unmanned servicing system to a manned system will require a docking adapter which allows the smaller docking mechanism of the manned system to dock with the larger unmanned docking port. Figure 3.5-5 illustrates this adapter. It essentially consists of two concentric docking mechanisms. The inner system is approximately five feet in diameter and is compatible with the space shuttle docking system. The outer mechanism is approximately seven feet in diameter and mates with the common support modules of the geosynchronous platforms. As previously stated, the large diameter on the common support modules is required for articulation of the interchange mechanism during auto-remote servicing operations.

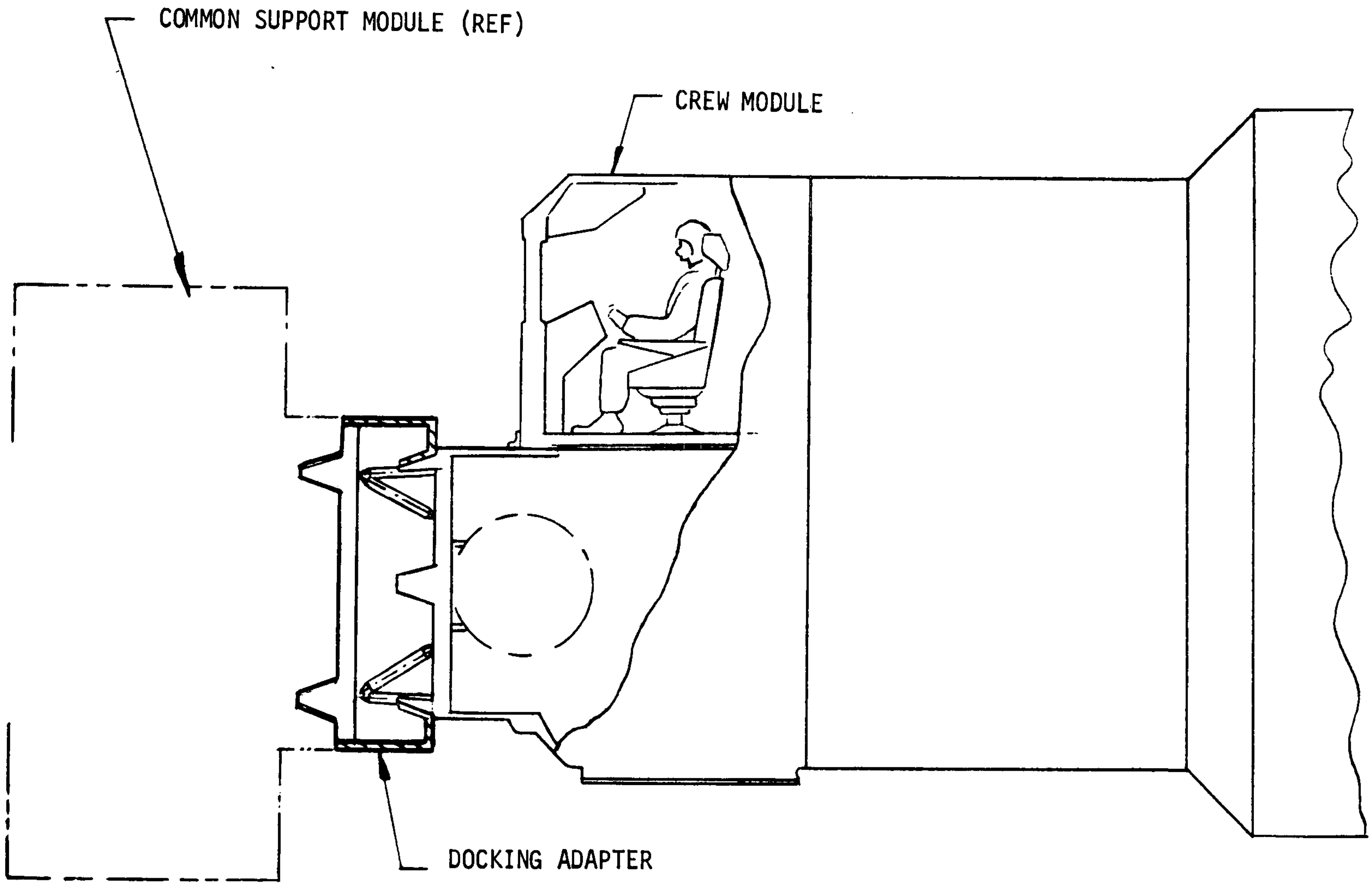

Figure 3.5-5. Dual Diameter Docking Concept

4.0 DATA RELAY PLATFORM

The configuration syntheses for platforms that primarily are for purposes of data relay are presented in this section. Functions included in this general category are Comsat, Domsat, Intelsat, TDRS, and navigation and traffic control.

Section 4.1 summarizes the pertinent international agreements and standards that govern the use of the frequency spectrum, radiation limits, and communication formats. Ground station characteristics are hypothesized (antenna, transmitter, receiver) to permit communication link calculations, signal margin determination, and the equipment complement on the platforms (antenna, transmitter, receiver).

Data relay platforms that fulfill the Comsat, Domsat, and navigation and traffic control requirements reflected in the baseline traffic model are synthesized in Section 4.2. Essentially, the only difference between the platforms for the four global regions is the number and type of transponder assemblies and antennas. A concept for real-time reallocation of channels via ground control is also developed in this section.

The TDRS functions are identical for the two traffic models. The new traffic model reflects the inclusion of foreign as well as U.S. TDRS's. The synthesis of the platform is presented in Section 4.3.

The proliferation of Intelsat and Domsat satellites in the new traffic model required a separate synthesis of a series of data relay platforms. Because of EM considerations, it is mandatory that multiple platforms be provided for each global region. Just Intelsat functions alone required two platforms for each region. The data relay platform accommodation and configuration concept for the new traffic model is presented in Section 4.4.

The unique navigation and traffic control platform required by the new traffic model is synthesized in Section 4.6. The emplacement requirements of each element of the four constellations precludes combining this function with other platforms. Both types of platforms are functionally the same as the corresponding satellites in the traffic models. Repackaging of equipment is accomplished to maximize commonality across the spectrum of geosynchronous elements and facilitate on-orbit servicing.

4.1 OPERATIONAL REQUIREMENTS

Synthesis of data relay platforms must account for necessary performance of mission operations and when global in nature--as any space operation is--perform within any existing international regulations and standards. Mission operations are supported by ensuring the necessary space and ground terminal characteristics that result in providing the telecommunications traffic needs with radiated power levels and receiver characteristics that result in high-quality (sufficient signal-to-noise ratios) signals to the users. At the same time, these operations must be performed in accord with national and international regulations and standards developed for the mutual benefit of all users. Recommended regulations and standards have been developed under the sponsorship of the International Telecommunications Union to protect other space operational missions and ground services from mutual interference problems. These factors and their relation to design synthesis of data relay platform operational systems are discussed in this section.

REGULATIONS AND STANDARDS

International Telecommunications Union (ITU)

Communications systems must be technically compatible with one another if they are to be interconnected and, since radio transmissions cannot be confined, collaboration is necessary to avoid interference. For these basic reasons, there exists the International Telecommunications Union (ITU), an agency of the United Nations. The specific functions of the ITU that directly affect the operation of any space telecommunications systems are:

1. Allocation of the radio frequency spectrum and registration of radio frequency assignments.

2. Coordination of efforts to eliminate harmful interference between radio broadcasts and to improve the use made of the radio frequency spectrum.

3. Undertake studies, make regulations, adopt resolutions, formulate recommendations and opinions, and collect and publish information concerning telecommunications for associate members.

To perform these functions, ITU has created a complex institutional structure. Three elements of the structured organization are as follows.

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1. Administrative conferences that meet at intervals of several years to revise regulations allocating radio frequencies and establishing procedures for international communications by radio.

2. Technical International Consultative Committee for Radio (CCIR) (comprised of plenary assemblies, periodically convened study groups and full-time specialized secretariats) that makes recommendations on technical specifications.

3. An International Frequency Registration Board (IFRB) that maintains a master register of frequencies and status of the use and its conformance to regulations.

World Administrative Radio Conference (1971)

The last World Administrative Radio Conference (WARC) convened in 1971 for the purpose of reviewing radio frequency allocations for space telecommunications. The results of these frequency allocations are covered in Volume III, Section 5.3. The 1971 conference raised the allocations of radio frequencies to 275 GHz, assigning many bands in the range from 1 GHz up for space use. Several band assignments are of particular significance for use in data relay platform operations. These are listed in Table 4.1-1. Higher frequency bands (to 275 GHz) have been allocated. These were, however, not considered applicable due,to technology limitations, for use in this study. Sufficient bandwidth is available in the Domsat/Intelsat bands (where the heaviest traffic density exists) of C, K_{LO}, and K_{HI} to accommodate the service requirements with a reasonable number of platforms.

Consultative Committee of International Radio (CCIR) Recommendations

Several CCIR recommendations impact the design requirements for fixed satellite communications link operations. Recommendations on the allowable levels of interference between adjacent satellite systems and between satellite systems and terrestrial communication links are of major importance.

Adjacent satellite interference limits and the method of calculating the levels have been fully discussed in Volumes III and IV. As these sections pointed out, adjacent satellite spacing for C-band operation is limited to a minimum of 4.6 degrees to maintain allowable interference levels. In both the baseline and the new traffic models, a minimum spacing of 5 degrees was maintained by utilizing high capacity data relay platforms.

Flux Density

CCIR has adopted regulations to protect satellite systems as well as terrestrial systems from mutual interference where frequency bands are shared.

CCIR recommendation 406-2 (Reference 4-1) for terrestrial systems to satellite systems limits terrestrial power levels of ground stations to 55 dBw. It further recommends that no ground station antenna point within 2 degrees of the direction to geostationary orbit.

Table 4.1-1. Data Relay Platform Frequency Allocations

Frequency (MHz/GHz)	Band	Bandwidth (MHz)	Link	Use
136 to 138 MHz	VHF	2	Space-to-earth	TDRS
148 to 149.9	VHF	1.9*	Earth-to-space	TDRS
401 to 402	UHF	1.0	Space operations (telemetry)	TDRS
1535.0 to 1558.5	L	23.5	Space-to-earth	Aircraft, maritime navigation and traffic control
1636.5 to 1660	L	23.5	Earth-to-space	
2020 to 2120	S	95	Earth-to-space	TDRS
2200 to 2290	S	95	Space-to-earth	TDRS
3.7 to 4.2 GHz	C	500	Space-to-earth	Domsat/Intelsat
5.925 to 6.425	C	500	Earth-to-space	Domsat/Intelsat
10.95 to 11.2	K_{LO}	500	Space-to-earth	Domsat/Intelsat
11.45 to 11.7	K_{LO}	500	Space-to-earth	Domsat/Intelsat
12.5 to 12.75	K_{LO}	500	Earth-to-space	Domsat/Intelsat
14.25 to 14.5	K_{LO}	500	Earth-to-space	Domsat/Intelsat
13.25 to 14.2	Ku	950	Earth-to-space	TDRS
14.4 to 15.35	Ku	950	Space-to-earth	TDRS
17.7 to 21.2	K_{HI}	3500	Space-to-earth	Domsat/Intelsat
27.5 to 31.0	K_{HI}	3500	Earth-to-space	Domsat/Intelsat

*Limited to transmission bandwidth of $\pm$15 kHz for space telecommand.

The major impact on satellite system design is on the flux density limits for satellite illumination of earth. Several CCIR recommendations apply to flux density limitations. They include limits for operations from 1 to 10 GHz (358-1, 387-1)* and above 10 GHz (Report 450)*. These recommendations define limits and calculations for various earth arrival angles. For use in this study, Figure 4.1-1 shows the tabulation of flux density limits for the various bands of interest. These describe the maximum limits of density under all conditions and correspond to the most recent guidelines.

*From Reference 4-2

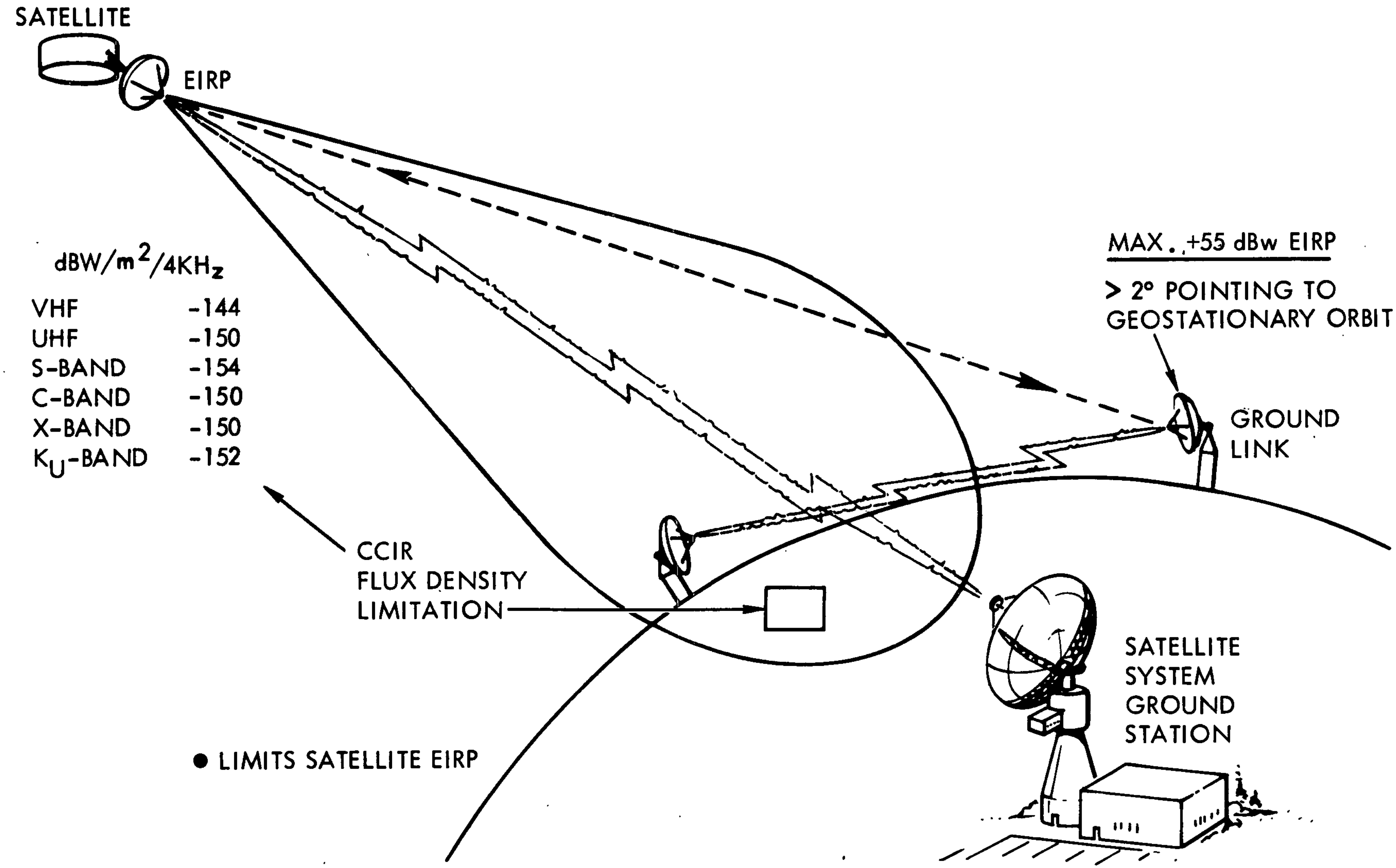


Figure 4.1-1. CCIR Power Limitations

In order to define the limits of EIRP for the different frequency bands, the following formula was used:

$$\text{Flux density*} = \frac{\text{EIRP} \cdot 4 \times 10^3}{(4\pi R^2 \cdot \text{Bw}} \times \text{dBw/m}^{-2}/4\ \text{kHz}$$

where EIRP = effective radiated power
R = distance from radiating element to earth
Bw = bandwidth in Hz

*Flux density defined by CCIR is in terms of dBw per meter squared in a 4 kHz band.

This was derived from the approximation of carrier flux density:

$$\text{Carrier flux density} = \frac{\text{EIRP}}{4\pi R^2}$$

and taking into account CCIR measurement within 4 kHz bandwidth and the actual bandwidth spread of the signal (Bw). In dB notation this becomes:

$$\text{FD} = \text{EIRP} - \text{Bw} - 126\ \text{dBw/m}^{-2}/4\ \text{kHz}$$

Using this formula, EIRP limits were calculated and are listed in Table 4.1-2.

Table 4.1-2. EIRP Limits for CCIR Flux Density Maximum Level

Frequency Band	EIRP (dBw)	Minimum Spread Bandwidth	Use
VHF	30.0	63 kHz	TDRS
UHF	30.0	250 kHz	
S	41.0	8 MHz	
X	48.0	16 MHz	
Ku	52.0	63 MHz	
C	51.6	36 MHz	Domsat
K_{LO}	49.6	36 MHz	Intelsat
K_{HI}	49.6	36 MHz	

These limits place no real constraints on data relay operations.

DESIGN CRITERIA

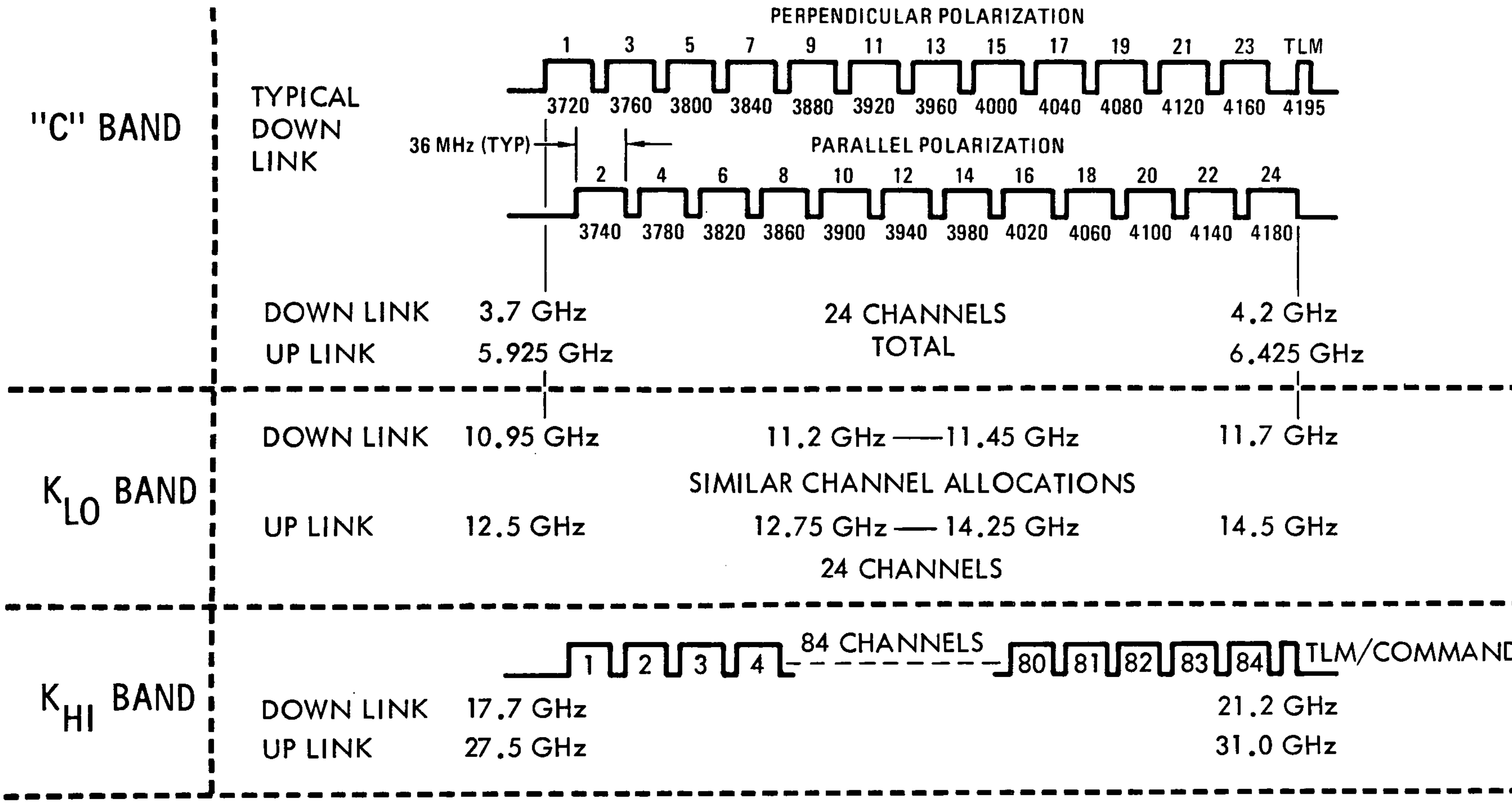
Data Relay Function

Design of a data relay communication link via a geostationary platform involves consideration of many ground and satellite terminal parameters. Design criteria are based on providing a signal at the user terminal with needed quality--or signal-to-noise ratio. The quality signal is produced when the received signal level at the user terminal is high enough under all levels of environment--both natural and man-made--to result in the required signal-to-noise ratio. The uplink ground-to-platform terminal, the platform transponder characteristics, and the platform-to-ground terminal downlink all contribute degradation of the original signal-to-noise ratio. The platform characteristics of (1) receiver system G/T (antenna gain, receiver noise temperature ratio) and (2) transmitter system EIRP (effective isotropic radiated power) are the critical design parameters. Ground system characteristics can more easily be adapted than spaceborne platform equipment. Large, heavy antennas, cooled low-noise receiver systems, and high-power transmitters can all be implemented on the ground within reasonable economic limitations. Radiated power levels of space equipment are limited by technology for the applicable frequency and by level of power input consistent with a reasonable power generation source. Antenna size and its gain must be traded off against beamwidth, desired coverage, and stability limitations. Receiver system noise figure is limited by the technology for uncooled receiver systems.

After the baseline design criteria for type of transmission, numbers of channels and bandwidth per channel are determined, the link parameters may be established based on the above-mentioned factors. For data relay functions, the frequencies and numbers of channels were established to meet traffic demands. The data relay platform requirements sections of Volume IV defined these criteria. Use of C-, K_{LO}- and K_{HI}-band was established for both the baseline and new traffic model.

Both the baseline and the new traffic model platforms used dual orthogonal polarization at C- and K_{LO}-bands. As previously discussed, use of K_{HI}-band single polarization was used in baseline platforms and dual polarization for the new traffic model. The resultant maximum single usage total of 132 - 36 MHz channels for the baseline model, and 216 channels for the new traffic model, were sufficient to allow a feasible number of platforms in each region. Figure 4.1-2 illustrates how these channels are arranged for the baseline traffic model platform. For K_{HI} dual polarization, the overlapping channel frequencies are centered in the same manner as for C-band, thus providing 168 channels instead of 84.

Figure 4.1-3 helps one to understand the tradeoffs associated with platform design criteria. An interrelationship exists between the platform EIRP and the ground antenna receiver system. It was determined for the traffic distribution that a 2-degree platform antenna beamwidth is a feasible minimum consistent with the necessary area coverage at K_{HI}-band. An antenna of 2-foot diameter satisfies this requirement and provides an antenna gain of approximately 38 dB. Examination of the ground antenna size indicates a 30-foot-diameter unit with a gain of 62.7 dB and a beamwidth of 0.14 degree is feasible both from an



- CHANNEL CAPACITY
  - 36 MHz BANDWIDTH
    - ONE FM COLOR TV SIGNAL
      OR
    - ≅ 1200 VOICE CHANNELS
    - ≅ 36 Mbps DIGITAL DATA

Figure 4.1-2. Frequency Allocation/Usage

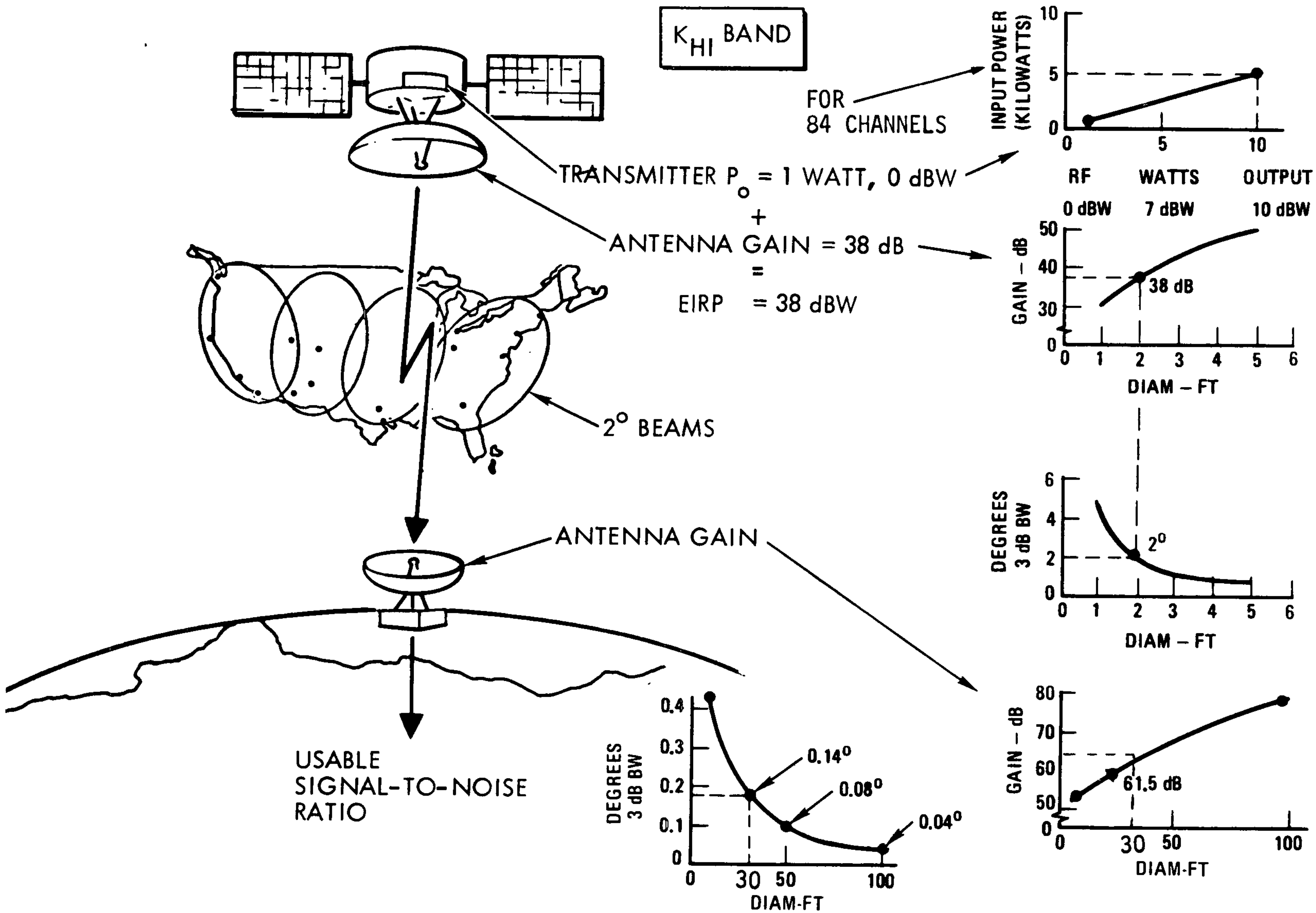


Figure 4.1-3. System Design Interrelationship

economic fabrication standpoint and compatibility with pointing and stability performance of the platform. With a reasonable ground receiver noise temperature (60 K) and allowance for rain margin, a platform per channel RF transmitter power output of one watt is sufficient to provide usable signal-to-noise ratios. One-watt-per-channel power output requires approximately 480 watts dc input power for 84 channels. As indicated, any higher power not only requires larger input power, but becomes more difficult to generate at these frequencies from a state-of-art standpoint. A similar analysis was performed for the other frequency bands.

The design criteria that resulted from this analysis are shown for each frequency band in Table 4.1-3.

Table 4.1-3. Data Relay Platform Design Criteria

Frequency Band	Platform Characteristics				Ground Antenna System		
	Power Output per Channel (watts)	Antenna					
		Size (ft)	Gain (dB)	Beamwidth HPBW* (degrees)	Size (ft)	Gain (dB)	Beam-width HPBW* (degrees)
C	4.0	3.5	29.8	5.0	60.0	54.7	0.3
K_{LO}	1.0	3.0	37.8	2.0	30.0	58.2	0.2
K_{HI}	1.0	2.0	38.3	2.0	30.0	62.7	0.14

*Half-power beamwidth

Margin calculations were performed for each of the frequency bands. Ground receiver noise temperatures and rain margins were assumed for each band as shown in the following:

	Receiver Noise Temperature (degrees K)	Rain Margin (dB)
C-band	45.0	0.5
K_{LO}-band	55.0	6.0
K_{HI}-band	60.0	12.0

Table 4.1-4 illustrates a typical margin calculation at C-band. Figure 4.1-4 shows a summary of all margin calculations and indicates the overall available margin under the worst environmental conditions that include rain and low (5-degree) mask angle for atmospheric absorption. This indicates that K_{HI}-band will have a small percentage of yearly interruptions at certain geographical locations unless diversity ground reception is used. It was felt that this interruption percentage was not sufficient to attempt spaceborne power increases of at least four-fold. In geographical areas where heavier rains occur, either diversity or larger size ground antennas are possible.

Table 4.1-4. C-Band Margin Calculations
(4 GHz Downlink Margin)

1.	Transmitter power in 36 MHz channel	+6.0 dBw
2.	Transmitting circuit losses	-1.6 dB
3.	Transmitting antenna gain (5-degree beam/1.05 m)	+29.8 dB
4.	EIRP	+34.2 dBw
5.	Polarization loss	-0.1 dB
6.	Space loss at 20.3 GHz at 35,860 km	-195.6 dB
7.	Sum of losses (items 5 and 6)	-195.7 dB
8.	Receiving antenna gain (60 ft/18.28 m)	+54.7 dB
9.	Total received signal power (items 4, 7 and 8)	-106.8 dBw
10.	Receiving noise spectral density at T_e = 45°K	-212.1 dBw/Hz
11.	Noise bandwidth at threshold (36 MHz)	+75.6 dB-Hz
12.	Threshold C/N required	+10.0 dB
13.	Threshold power (items 10, 11 and 12)	-126.5 dBw
14.	C/N performance margin (items 9 - 13)	+19.7 dB

Ground flux densities were calculated for each band and are shown on the chart of Figure 4.1-4. As indicated, flux densities are well within the CCIR requirements.

In summary, the design criteria to be used for the data relay equipment of the platform are outlined on Table 4.1-3.

Navigation and Traffic Control Function

An Aerosat-type navigation and traffic control system is utilized for platform design criteria of the baseline traffic model. The following functions are implemented.

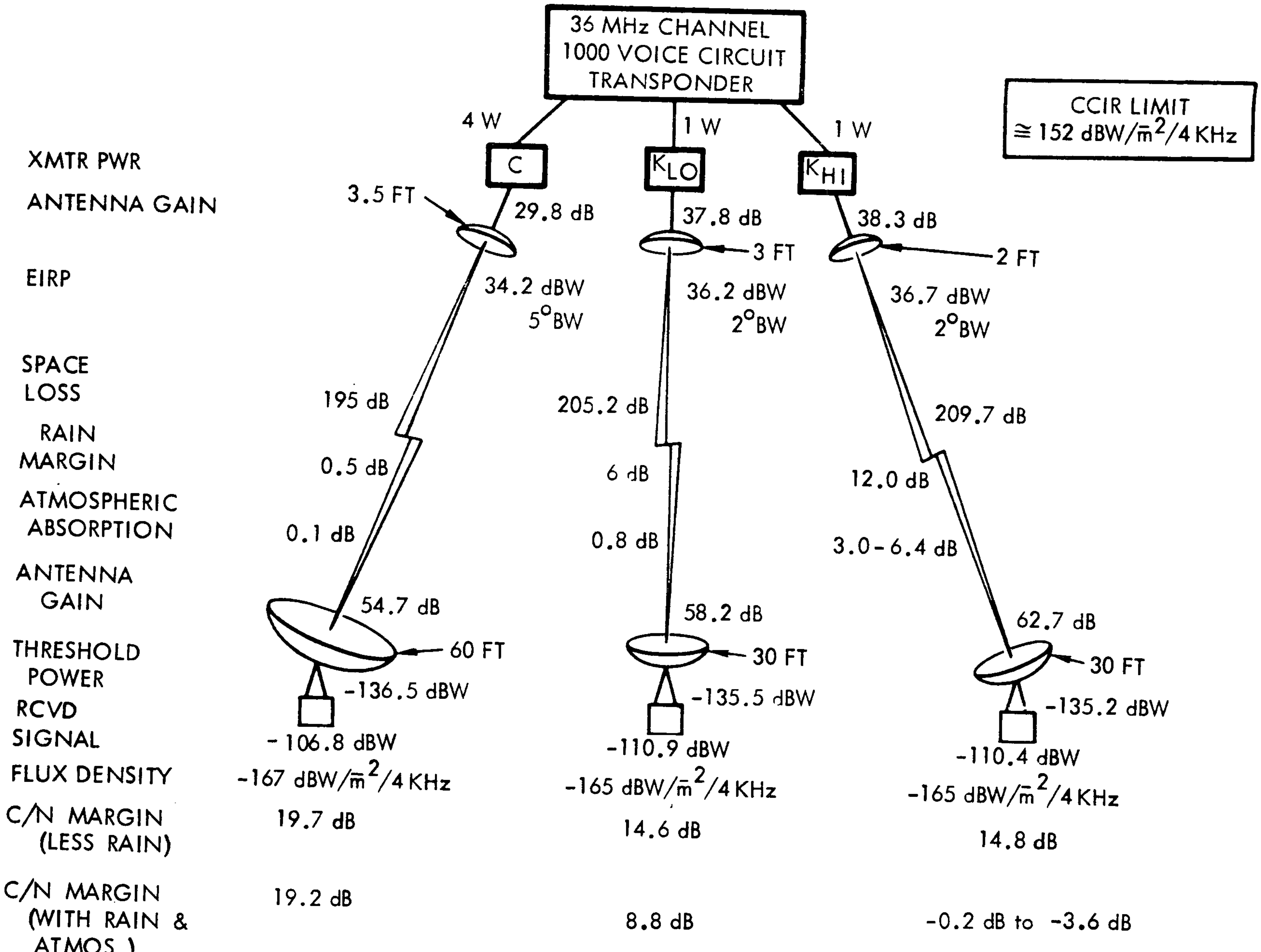


Figure 4.1-4. Data Relay Platform

Functions	Link
1. General communications between ground and aircraft	Two-way voice or digital data
2. Surveillance of aircraft by ground control	Digital data relay of aircraft equipment-derived navigation data to ground
3. Aircraft control	Voice or digital data link from ground to aircraft

A system has been conceived by the FAA to minimize the economic effect on the world-wide aircraft fleet. As pointed out in the second function, navigation information will be available from on-board inertial navigation system (INS) equipment. Automated techniques can be utilized to transmit these data on a regular basis over the digital data link. The aircraft-to-space platform transmission can be made by use of existing TACAN equipment now utilized by all commercial aircraft. Thus, very little new aircraft equipment is needed. Voice can also be multiplexed over the same link. Return voice and/or data are transmitted back to the aircraft over a frequency in L-band allocated for navigation uses.

Figure 4.1-5 illustrates the major design parameters of this system. L-band is used for the aircraft/platform link, and C-band for the ground/platform link. The frequency bands are as follows:

	Uplink	Downlink
Aircraft - L-band	1.025 GHz to 1.150 GHz	1.54 GHz to 1.66 GHz
Ground	Use Domsat data relay C-band channel.	

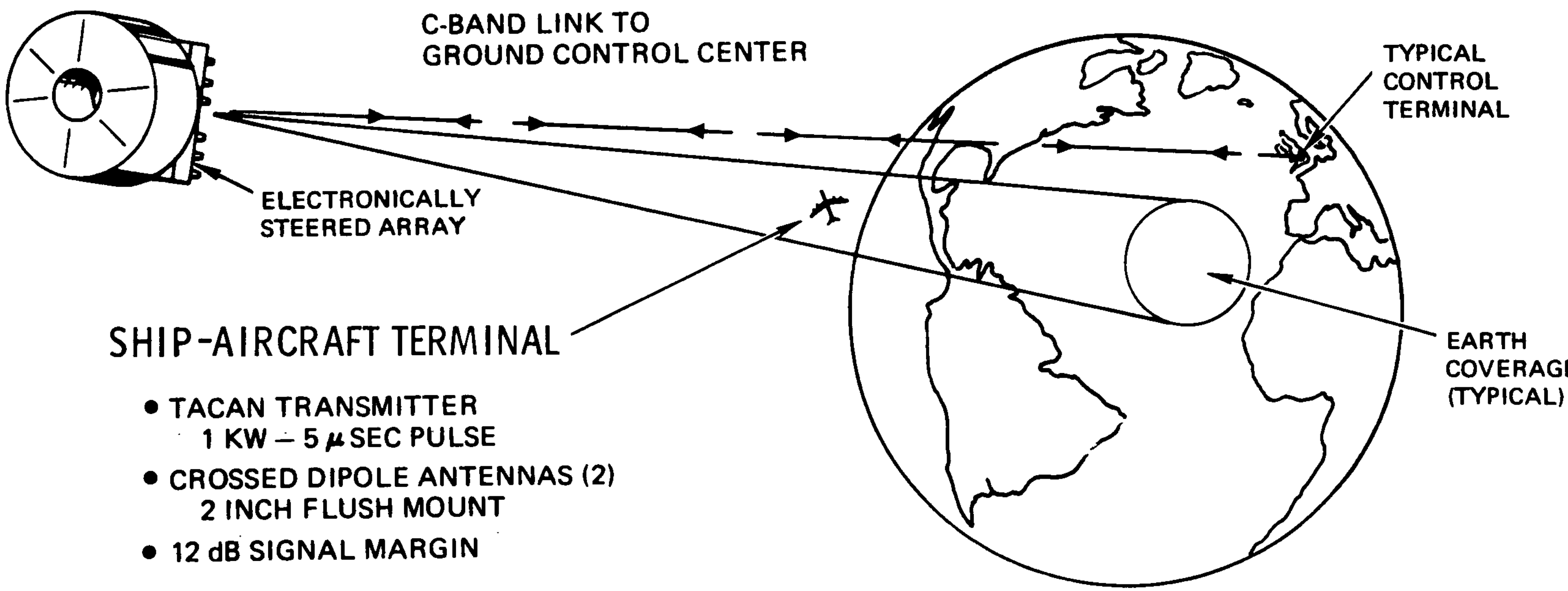
In order to perform the platform-to-aircraft communications with feasible spaceborne power levels, a high-gain antenna is needed at L-band. This, however, limits the beamwidth and the coverage of the antenna. To take advantage of this antenna gain and narrow beamwidth antenna, a TDMA system using a phased array antenna on the platform is proposed. By using electronic scanning of the phased array and assigning time slots to aircraft in the various locations within the scanning pattern, each aircraft can be contacted for sufficient time to provide the necessary communications transfer.

SATELLITE TERMINAL

- 50 WATT TRANSMITTER
- ARRAY ANTENNA
  29 dB GAIN
  6° BEAMWIDTH
- 9 dB SIGNAL MARGIN

SYSTEM CHARACTERISTICS

- UPLINK 1.025 GHz
- DOWNLINK 1.54 GHz
- 250 KBPS DATA RATE
- TDMA PCM VOICE/DATA



SHIP-AIRCRAFT TERMINAL

- TACAN TRANSMITTER
  1 KW – 5 μSEC PULSE
- CROSSED DIPOLE ANTENNAS (2)
  2 INCH FLUSH MOUNT
- 12 dB SIGNAL MARGIN

Figure 4.1-5. Navigation and Traffic Control Baseline Model

Based on these characteristics and the use of feasible receiver system noise temperatures, a set of link margin calculations were made to assure suitable operation. The characteristics used for these calculations are listed below:

	Platform Transponder	Aircraft Transponder
Transmitter power	50 watts	1000 watts
Antenna gain		
Receive	19.0 dB	2.0 dB
Transmit	29.0 dB	2.0 dB
Receiver noise temperature	600 K	300 K
Signal data rate	250 kbps	250 kbps
Frequency		
Transmit	1.54 to 1.66 GHz	1.025 to 1.15 GHz
Receive	1.025 to 1.15 GHz	1.54 to 1.66 GHz

Tables 4.1-5 and 4.1-6 detail the margin calculations for these links. Sufficient margin is shown in both cases.

Table 4.1-5. Downlink (Platform-to-Aircraft) Margin Calculations

Downlink to Aircraft (1540 MHz)		
1.	Transmitter power (50 watts)	+17.0 dBw
2.	Transmitting antenna gain	+29.1 dB
3.	Satellite-to-aircraft space loss maximum	-187.4 dB
4.	Transmitting losses	-0.5 dB
5.	Polarization loss	-0.5 dB
6.	Receiving antenna gain	+2.0 dB
7.	Receiving circuit losses	-0.5 dB
8.	Net losses (2 + 3 + 1 + 5 + 6 + 7)	-157.8 dB
9.	Received signal power	-140.8 dB
10.	Receiver noise temperature, 300 K	+24.7 dB
11.	Boltzmann's constant	-228.6 dBw
12.	Receiving noise density	-203.9 dBw/Hz
13.	Bandwidth at 250 kbps	+53.9 dB-Hz
14.	Received noise per unit bandwidth	-150.0 dBw
15.	Energy per bit/noise per unit bandwidth	+9.2 dB
16.	Required E_B/N_O for BER = 1×10^{-4}	+8.5 dB
17.	Margin (15 - 16)	+0.7 dB

G

Table 4.1-6. Uplink (Aircraft-to-Platform)
Margin Calculations

Uplink to Satellite (1025 MHz)		
1.	Transmitter power (1000 watts)	+30.0 dBw
2.	Transmitting antenna gain	+2.0 dB
3.	Aircraft-to-geosynchronous space loss, maximum	-183.8 dB
4.	Transmitting losses	-0.5 dB
5.	Polarization loss	-0.5 dB
6.	Receiving antenna gain	+19.0 dB
7.	Receiving system losses	-0.5 dB
8.	Net loss (2 + 3 + 4 + 5 + 6 + 7)	-164.3 dB
9.	Received signal power	-134.3 dBw
10.	System noise temperature, 600 K	+27.8 dB-deg K
11.	Boltzmann's constant	-228.6 dB/deg K/Hz
12.	Signal bandwidth at 250 kbps	+53.9 dB-Hz
13.	Received noise power	-146.9 dBw
14.	Received signal power	-134.3 dBw
15.	Signal-to-noise ratio	-12.6 dB

4.2 DATA RELAY PLATFORM SYNTHESIS FOR BASELINE TRAFFIC MODEL

Design of the data relay platform for the four global regions must include consideration of orbit-earth geometry, geographic peculiarities of each region, international boundaries, dynamic traffic requirements, and equipment weight, power, and volume constraints. All of these factors must be evaluated in light of the operational requirements identified in Section 4.1.

In this section of the report a platform antenna configuration for each region is defined that provides adequate national and international coverage. The approach to achieve area coverage within the 5-degree mask angle of each region was as follows: A computer program was composed to plot antenna beam ground intercept foot prints on a cylindrical projection map of the earth. The program permitted selection of the beamwidth, platform longitudinal location, and antenna boresight direction. Beamwidths and boresight directions were varied until adequate coverage of all countries (Domsat) and inter-country links were obtained. Primarily, circular beam patterns were assumed. However, the ground intercept projections appear elliptical and other assorted shapes because of the geometry of the earth-synchronous orbit relationship.

For the regions to be discussed, each is assigned two platforms of equal capacity longitudinally spaced 10 degrees apart. This avoids solar eclipse outage problems on the platforms and permits double use of all frequencies due to the platform physical separation and attendant earth station antenna pointing selectivity. The requirements to be discussed thus represent only one-half of the total requirement. The beam patterns plotted are for half-power (3 dB) countours. Link operation beyond these limits is possible at reduced margins of performance or at the same margin using larger earth antennas. Gain calculations indicate that link integrity can be maintained by use of 60-foot-diameter antenna reflectors for C-band ground stations and 30-foot-diameter reflectors for K-band.

A given regional platform covers both Domsat- and Comsat-type functions for all areas within the region. Comsat coverage for a specific country may involve platforms in two different regions to ensure its link to other communicating countries that are located in opposite directions.

Each regional platform is also equipped with an 8-3/4-foot rectangular phased array steered electronically with a 6-degree beam that can be aimed to cover any sector of the 5-degree mask area for the region. This system provides the additional capability for the platform necessary for navigation/air traffic voice and data service.

Packaging concepts for the platform mission equipment are derived in this section. The concepts include a technique for channel reallocation via ground commands.

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Layouts of the data relay platforms are presented. The replacement schedule of expendable satellites by the data relay platform is also defined.

REGION I DATA RELAY PLATFORM REQUIREMENTS

Global Region I encompasses the Pacific Ocean area. It includes the U.S., Canada, and Mexico on its eastern extremity and Japan, Russia, China, Southeast Asia, and Australia on its western extremity.

Based upon the satellite inventory in the baseline traffic model, 12 Domsat and 12 Comsat channels are required per platform for servicing this region. Manipulation of the boresight aim points of 5-degree,C-band antenna beams resulted in providing adequate coverage with five beams. Figure 4.2-1 illustrates the coverage and beam aim points.

C-band channels C_1 through C_{12} were allocated for Domsat purposes. C_1, C_2 and C_3 were assigned to Japan; C_4, C_5,and C_6 were allocated to the Phillipines and East India; Australia was assigned C_7, C_8,and C_9; New Zealand and some of the Pacific islands were allocated channels C_{10}, C_{11}, and C_{12}.

The Comsat requirements were fulfilled with C-band channels C_{13} through C_{24}. One of the primary drivers in the selection of the antenna beam characteristics was to provide multiple paths for inter-country communications. Note the location of the beam that encompasses Canada, the U.S., and Mexico. This facilitates alternate communication paths between the Western Hemisphere and the Far East. Some of the key cities in the Far East that were intentionally included in the beams to provide Comsat service were Vladivostok, Russia; Peking/Shanghai/Canton, China; Bangkok, Thailand; Rangoon, Burma; and Saigon, Viet Nam.

Two 3.5-foot-diameter parabolic reflectors with two offset feed horns were selected to produce the beams covering the Far East. A single feed horn on a separate parabolic reflector was assumed to obtain the beam illuminating the western extremity of the region. All antennas include dual polarization.

The entire 5-degree mask angle of the region is scanned by a 6-degree electronically steered beam from a phased array that is 8.75 feet by 8.75 feet. This concept will accommodate the data relay for the navigation and traffic control function.

In summary, each data relay platform in Region I will require the following complement of mission equipment:

- One 24-channel C-band transponder
- Two dual feed horn, dual polarized 3.5-foot-diameter parabolic reflector antennas
- One single feed horn, dual polarized 3.5-foot-diameter parabolic reflector antenna
- One L-band transponder
- One 8.75 x 8.75 L-band phased array antenna

- **REQUIREMENTS/PLATFORM**
  - DOMSAT 12 CHANNELS
  - COMSAT 12 CHANNELS
- **ACCOMMODATION CONCEPT**
  - DOMSAT $C_1 \rightarrow C_{12}$ (C-BAND)
  - COMSAT $C_{13} \rightarrow C_{24}$ (C-BAND)
- **EQUIPMENT COMPLEMENT**
  - C-BAND
    - 2- PARABOLIC ANTENNA
3.5 FT DIAMETER
DUAL FEED
DUAL POLARIZATION
    - 1- PARABOLIC ANTENNA
3.5 FT DIAMETER
SINGLE FEED
UNIPOLARIZATION

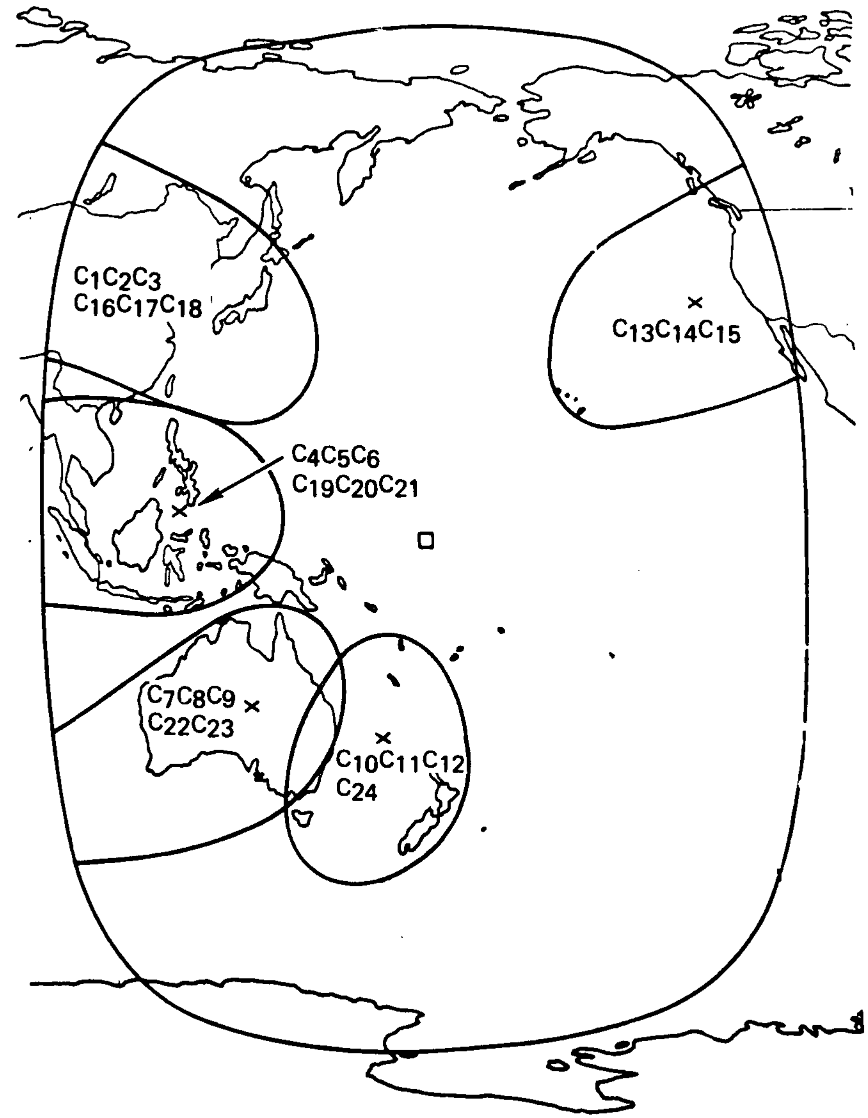


Figure 4.2-1. Region I Data Relay Platform System Characteristics

The optimum location for a Region I platform is 172 degrees east longitude. Because of the sun outage, two platforms are required, and they are spaced 10 degrees apart. Thus, the platforms are placed at 167 degrees and 177 degrees east longitude. Only a minor perturbation of the antenna patterns depicted in Figure 4.2-1 results from the offsets from the optimum point.

REGION II DATA RELAY PLATFORM REQUIREMENTS

Global Region II encompasses Europe, Asia, Africa, and Australia. Analysis of the baseline traffic model indicated that 18 channels of Domsat capability and 12 channels of Comsat capability per platform are required in this region. It was assumed that European Domsat functions were not included in this region. Inclusion of Europe in this region was for Comsat purposes only.

Because of the large land masses included in this region, it was highly desirable to achieve coverage with the wide-beamwidth C-band concept. Two-degree beamwidths associated with K-band systems would lead to a proliferation of antennas on the platform in order to achieve adequate coverage. A total of 30 channels are required in Region II, but only 24 channels are available in the C-band of the spectrum. Therefore, frequency reuse is required.

Frequency reuse on a single platform is practical if dedicated up-down-links are prescribed and the separation between antenna beams associated with the same carrier frequency is at least one beamwidth. Domsat functions can be readily accommodated in a frequency reuse scheme.

Figure 4.2-2 illustrates the optimized ground intercept patterns for Region II. Channels C_1 through C_{12} are allocated to the Domsat function. However, note that C_1 through C_8 are used twice. For example, C_5, C_6, and C_7 are used for Domsat in both the Middle East and Indonesia. Thus, 18 Domsat channel requirements can be achieved by only 12 C-band channels.

The Comsat channels, C_{18} through C_{24}, were arbitrarily assigned throughout the region. Beam boresights were selected to provide maximum international coverage. Communications between England and Japan; England and Melbourne, Australia; and Japan and Cape Town, South Africa are examples of the coverage extremities.

Use of dual feed horns and dual polarization permits a reduction in the required number of parabolic reflectors. Structural limitations require that a squint angle between feed horns be at least equal to the width of the desired beam and less than three beamwidths. For example, the beam illuminating Australia can use the same parabolic reflector as the one illuminating China. It could not be generated on the same reflector with the beam illuminating Southeast Asia (overlapping beam patterns) or the beam illuminating Russia (greater than three beamwidths away).

Because of the proximity of the central area of the region (India, Pakistan, Iran, Saudi Arabia) to adjacent beams, it was determined that multiple single feed horns would be required to obtain adequate coverage. Therefore, a single shaped beam antenna concept was selected to cover the

- **REQUIREMENTS/PLATFORM**
  - DOMSAT 18 CHANNELS
  - COMSAT 12 CHANNELS
- **ACCOMMODATION CONCEPT**
  - DOMSAT $C_1 \rightarrow C_{12}$ (C–BAND)
  - COMSAT $C_{13} \rightarrow C_{24}$ (C–BAND)
- **EQUIPMENT COMPLEMENT**

  C–BAND

  1- SHAPPED BEAM ANTENNA
  110 IN. X 60 IN.
  DUAL POLARIZATION

  3- PARABOLIC ANTENNAS
  3.5 FT DIAMETER
  DUAL FEED
  DUAL POLARIZATION

  1- PARABOLIC ANTENNA
  3.5 FT DIAMETER
  SINGLE FEED
  DUAL POLARIZATION

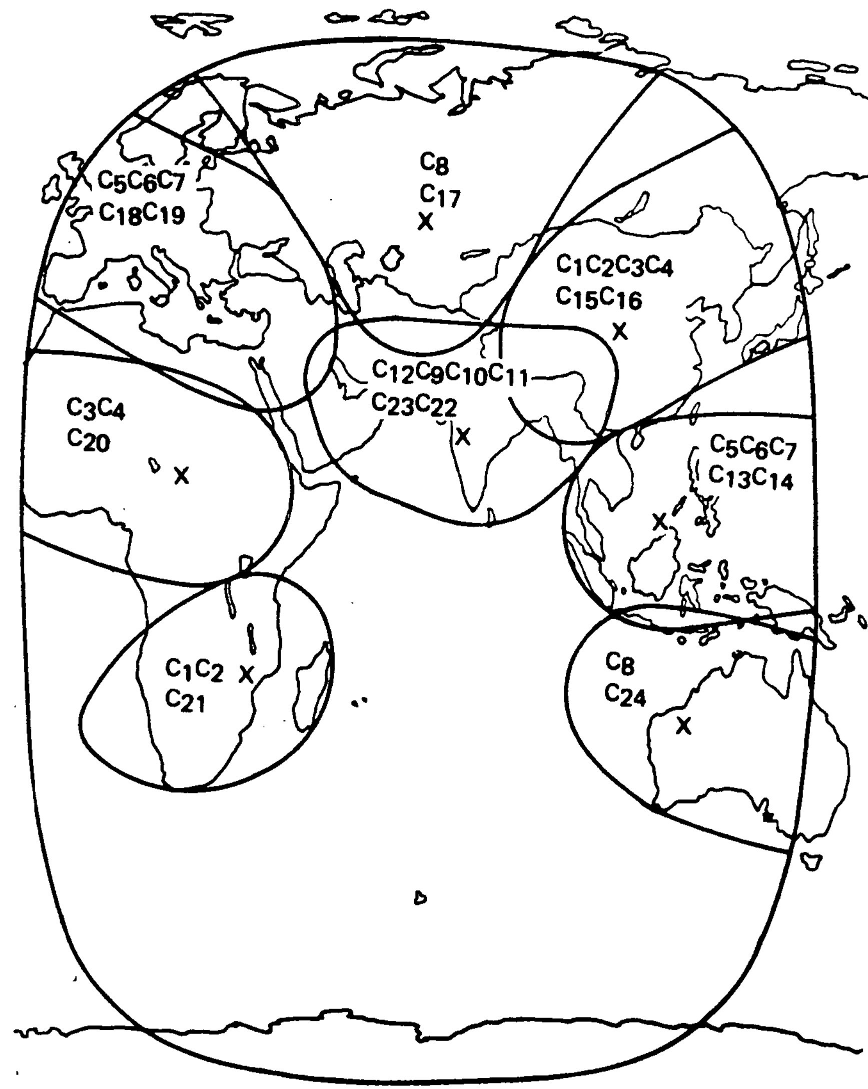


Figure 4.2-2. Region II Data Relay Platform System Characteristics

entire central area. It has a single feed horn with a reflector that is 110 inches long on the major axis and 60 inches long on the minor axis. Beam shaping is accomplished by spoiling the reflector true parabolic curvature and by feed offset.

As in Region I, the navigation and traffic control function is provided by an L-band transponder and an 8.75 by 8.75-foot phased array.

In summary, the mission equipment complement for each data relay platform for Region II is as follows:

One	24-channel C-band transponder
One	6-channel C-band transponder
Three	dual feed horn, dual polarized, 3.5-foot-diameter parabolic reflector antennas
One	single feed horn, dual polarized, 3.5-foot-diameter parabolic reflector antenna
One	single feed horn, dual polarized shaped antenna 110-inch major axis by 60-inch minor axis
One	L-band transponder
One	8.5 by 8.5-foot L-band phased array antenna

Because the C-band channel breakdown is 12 channels per polarization, a nominal packaging concept would be a minimum of 12 channels per assembly. Therefore, it is assumed that the 6-channel transponder listed above would be replaced by a 12-channel unit.

Each of the Region II platforms are positioned five degrees in longitude from the optimum point of 68 degrees east. No significant changes to the ground intercepts result from this shift from the optimum point.

REGION III DATA RELAY PLATFORM REQUIREMENTS

Europe, the Middle East, Africa, South America, and the eastern coastline of the United States and Canada are included in Region III. The requirements of each platform in this region expand to 66 channels. Forty-two channels are required for Domsat service; 24 channels are required for Comsat service. South American Domsat requirements will be fulfilled by Region IV platforms. Region II platforms provided the Domsat capability for the Middle East and Africa. Therefore, all 42 Domsat channels are assigned to Europe.

The required Comsat coverage in this region encompasses numerous countries and far-flung land masses. Therefore, it is highly desirable to accommodate this requirement with the wide-beam C-band system. Resorting to K-band would result in a proliferation of antennas on the platforms. A 24-channel Comsat requirement can be accommodated by using both C-band polarizations. Figure 4.2-3 illustrates the ground intercepts of an 8-beam C-band system.

- REQUIREMENTS/PLATFORM
  - DOMSAT 42 CHANNELS
  - COMSAT 24 CHANNELS
- ACCOMMODATION CONCEPT
  - DOMSAT $K_1 \rightarrow K_{42}$ (K_{HI} BAND)
  - COMSAT $C_1 \rightarrow C_{24}$ (C-BAND)
- EQUIPMENT COMPLEMENT
  - C-BAND
    - 4- PARABOLIC ANTENNAS
      3.5 FT DIAMETER
      DUAL FEED
      DUAL POLARIZATION
  - K_{HI}- BAND
    - 2- PARABOLIC ANTENNAS
      2 FT DIAMETER
      DUAL FEED
      UNIPOLARIZATION

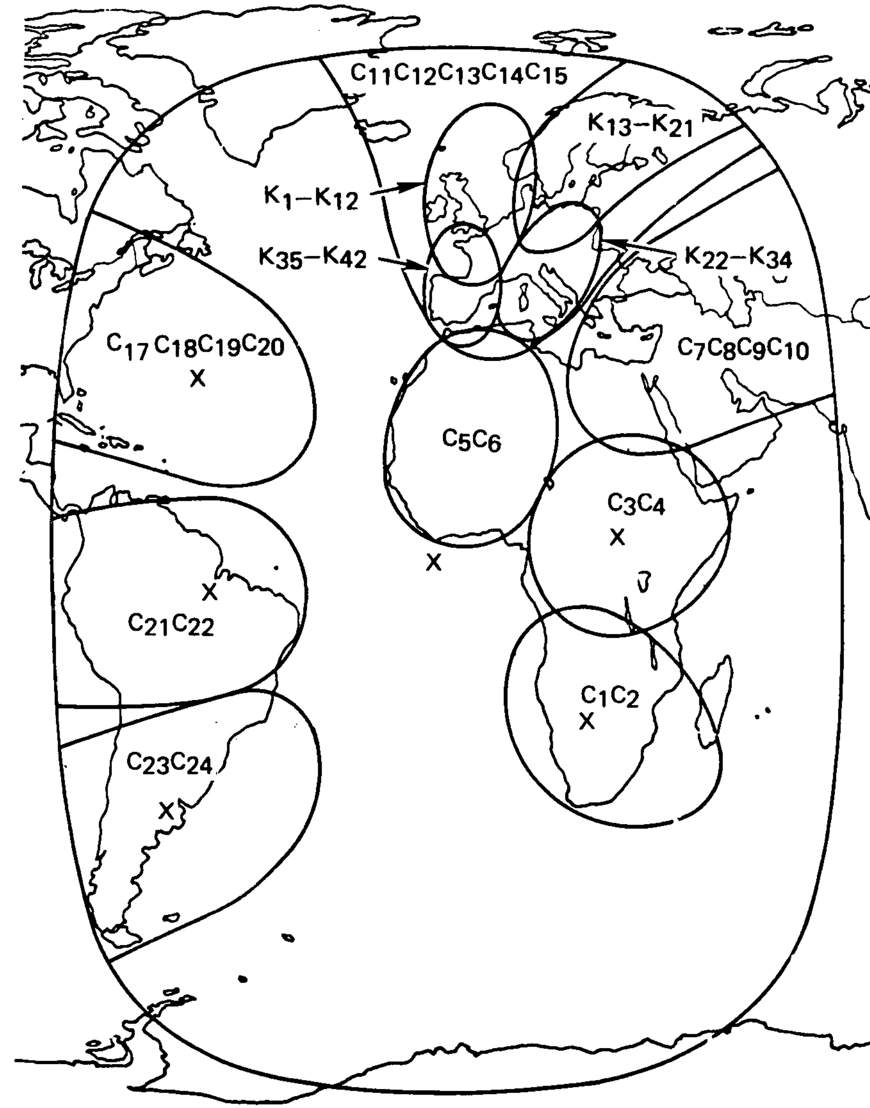


Figure 4.2-3. Region III Data Relay Platform System Characteristics

Use of the C-band system to achieve a 42-channel European Domsat capability is not feasible. The compact geographics of Europe preclude a K_{LO}-band frequency reuse concept. Only 24 channels are available in the K_{LO}-band. Therefore, K_{HI}-band was selected to provide the Domsat service for Europe. The ground intercepts of the K_{HI}-band beams are also shown in Figure 4.2-3.

A second reason for selecting K_{HI}-band for the European Domsat function is the number of sovereign nations in the area that have a rather high degree of technological capability. The possibility of an even greater demand for channels in the 1980 time frame can be readily accommodated with the K_{HI}-band system. With just unipolarization, K_{HI}-band can accommodate 84 channels.

The navigation and traffic control function is accommodated in the same manner as in Regions I and II, an L-band transponder with a phased array.

Use of dual feed horns on parabolic dishes result in the following total mission equipment complement for Region III data relay platforms:

One	24-channel C-band transponder
Four	dual feed horn, dual polarized, 3.5-foot diameter parabolic reflector antennas
One	42-channel K_{HI}-band trnasponder
Two	dual feed horn, unipolarized, 2.5-foot diameter parabolic reflector antennas
One	L-band transponder
One	8.5 by 8.5-foot L-band phased array antenna

Because of the inherent 84-channel capability on the K_{HI}-band and the potential increased demand for European Domsat service, an 84-channel system was substituted for the 42-channel system. Region III platform layouts reflect this substitution.

Each platform is positioned five degrees in longitude from the optimum Region III location of six degrees west longitude. The delta five-degree west placement will preclude direct North America-India communications through that particular platform. No other significant perturbations result from the platform offset.

REGION IV DATA RELAY PLATFORM REQUIREMENTS

The Western Hemisphere is encompassed in Region IV and is illustrated in Figure 4.2-4. All requirements for this region that were extracted from the baseline traffic model were for Domsat service only. Neither Comsat nor navigation and traffic control requirements were defined. During the synthesis of the Region IV platform, the requirements were arbitrarily expanded to include these functions. Comsat requirements were assumed to be six channels. The navigation and traffic control requirements were assumed to be the same as all other regions. They are: (1) periodic communications to the mask angle limits, and (2) capability to relay data up to 250 K bits per second.

Domsat requirements were defined as 144 channels per platform. Six were identified for Canada and 120 for the U.S. The remaining 18 were assigned to Central and South America. Even with the use of C-band 5-degree beams and a shaped beam, three beams were required to cover Central and South America. Six channels were arbitrarily assigned to each of these beams.

Because Canada and the United States are contiguous, frequency reuse in covering these two countries is not practical. The large land mass of Canada coupled with the relatively small required number of channels dictates the use of C-band for its Domsat functions.

Reserving six channels for Western Hemisphere Comsat functions leaves only 12 C-band channels for U.S. Domsat functions. Therefore, 108 channels must be provided by K-band systems. Both K_{LO} AND K_{HI}-band systems utilize a two-degree beam. Adequate coverage of the U.S. can be accomplished with four two-degree beams. It is assumed that the major portion of the U.S. Domsat traffic is teleconferencing and regional distribution centers will be used.

A shaped beam antenna is proposed for the C-band system illuminating the continental U.S. (CONUS). Total CONUS coverage is provided for the 12 C-band channels, which could be used for nationwide television broadcasts.

Alaska and Hawaii are accommodated by C-band beams also. Separate channel and transponders are not proposed for these two areas. As they are an integral part of the U.S. the same C-band channels serving CONUS service Alaska and Hawaii. This is accomplished by providing a microwave tap between the feed to the shaped beam antenna and the antenna illuminating Alaska and Hawaii. The concept is illustrated in Figure 4.2-5.

Shaped beams are also required to obtain effective coverage of Canada and Central America. Two five-degree circular beams provide adequate coverage of South America.

The six Comsat channels were arbitrarily subdivided among the five independent C-band beams--one to each beam except the U.S. beam which has two.

- **REQUIREMENTS/PLATFORM**

DOMSAT	144 CHANNELS
COMSAT	NONE SPECIFIED

- **ACCOMMODATION CONCEPT**

DOMSAT	K_1 - K_{84} (K_{HI} BAND)
	K_{85} - K_{108} (K_{LO} BAND)
	C_1 - C_{18} (C-BAND)
COMSAT	C_{19} - C_{24} (C-BAND)

- **EQUIPMENT COMPLEMENT**

C-BAND

3 - SHAPED ANTENNAS
110 IN. X 60 IN.
SINGLE FEED
DUAL POLARIZATION

2 - PARABOLIC ANTENNAS
3.5 FT DIAMETER
DUAL FEED
DUAL POLARIZATION

K_{HI}-BAND

2 - PARABOLIC ANTENNAS
3 FT DIAMETER
DUAL FEED
SINGLE POLARIZATION

K_{LO}-BAND

2 - PARABOLIC ANTENNAS
3 FT DIAMETER
DUAL FEED
DUAL POLARIZATION

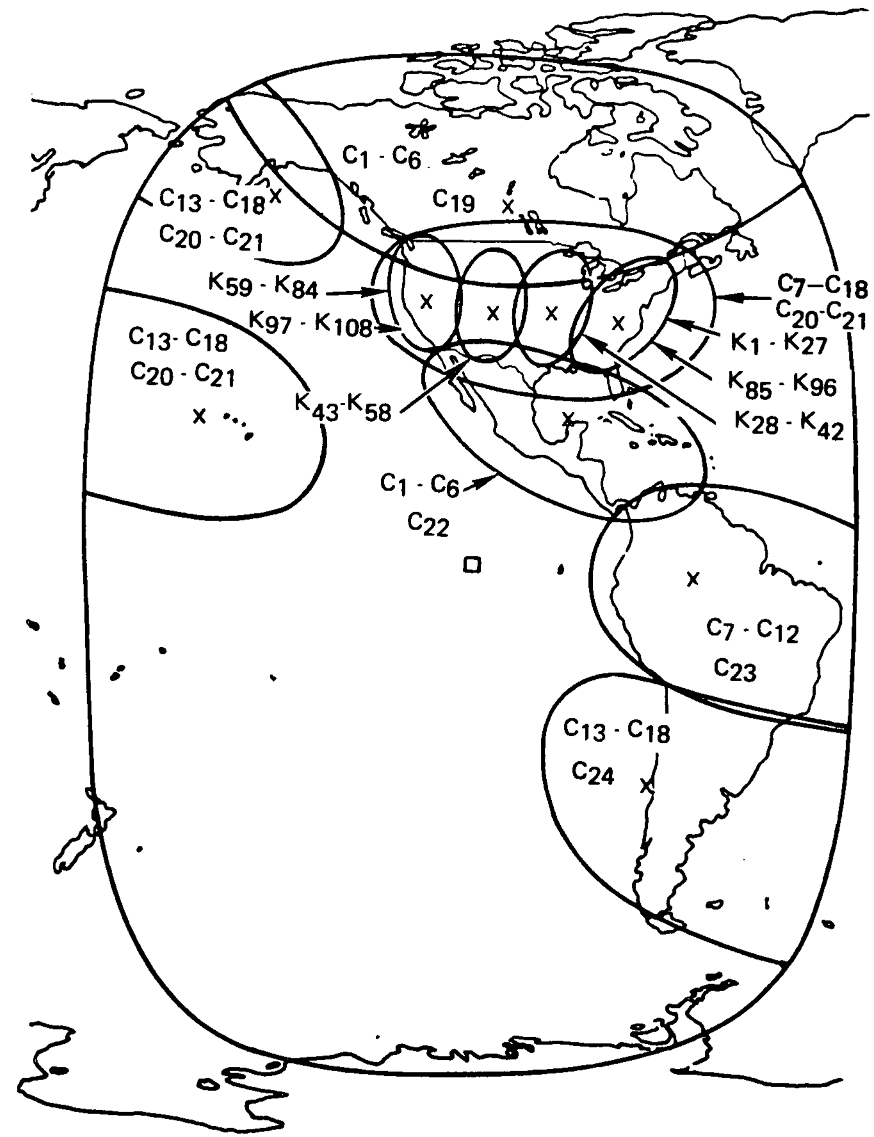


Figure 4.2-4. Region IV Data Relay Platform System Characteristics

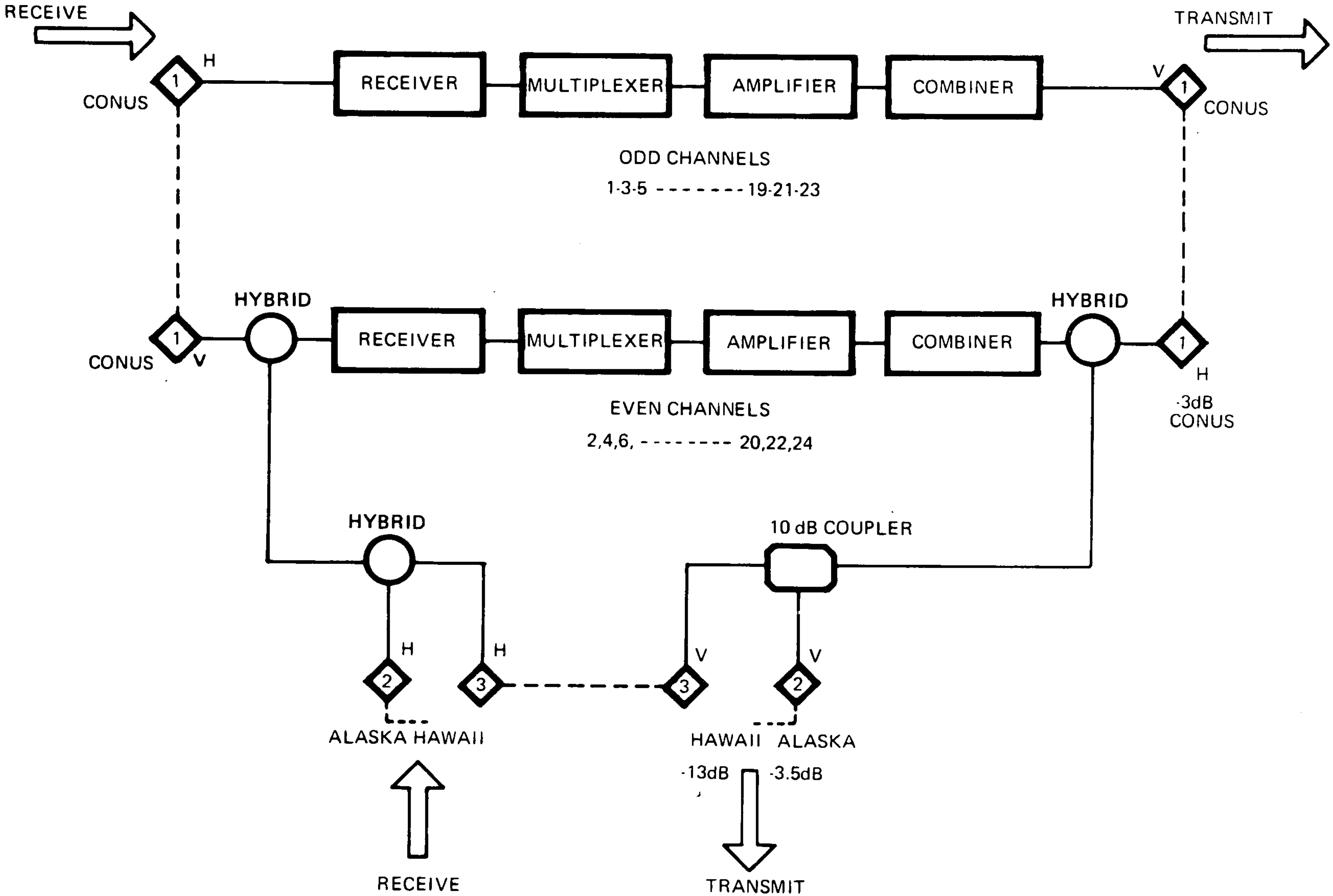


Figure 4.2-5. Alaska/Hawaii Transponder Feed Arrangement

The mission equipment complement of each Region IV data relay platform is as follows:

Two	24-channel C-band transponders
Three	single feed horn, dual polarized, shaped antennas 110-inch major axis by 60-inch minor axis
Two	dual feed horn, dual polarized, 3.5-foot-diameter parabolic reflector antennas
One	24-channel K_{LO}-band transponder
Two	dual feed horn, dual polarized, 3-foot-diameter parabolic reflector antennas
One	84-channel K_{HI}-band transponder
Two	dual feed horn, unipolarized, 2.5-foot-diameter parabolic reflector antennas
One	L-band transponder
One	8.5 by 8.5-foot L-band phased array antenna

The Region IV data relay platforms are to be placed at 115 degrees and 105 degrees west longitude. This provides a 10-degree separation to circumvent the sun outage problem. The 5-degree displacement from the optimum Region IV location (110 degrees west longitude) does not cause any significant perturbations in the regional coverage.

CHANNEL ASSIGNMENT FLEXIBILITY

The channel assignments in each region were arbitrarily made. If fixed assignments could be made, the equipment would be relatively simple. However, this is not considered practical. Accurately predicting the requirements of each country ten years in advance is impossible. Also, traffic demands within a region will fluctuate as a function of the time of day. Therefore, it was considered mandatory that some type of channel reallocation scheme be incorporated in the data relay platforms.

Trades were evaluated between Frequency Division Multiplexing (FDM) and Time Division Multiplexing (TDM) as basic techniques. These are discussed at length in the literature, and this study did not attempt to add information to that already available. Purely on the basis of initial design complexity, a system that would use TDM with multiple access for the users (TDMA) was rejected for consideration for platform systems. The number of channels that would require real-time sharing was not considered large enough to warrant the level of on-board detection, decoding, synchronism, and switching complexity required for TDMA.

Fixed Channel Assignment

An example of the trades considered includes a scheme for FDM with pre-fixed channel allocations permanently wired in the on-board system. Point-to-point coverage for CONUS provides a representative requirement on which to base the configuration. Using 84 channels available in the K_{HI}-band, a scheme can be devised to distribute the channels according to estimated traffic in the time span of intended service. One scheme is shown in Figure 4.2-6, which works with four 2-degree beams within which the 84 channels of the K_{HI}-band are allocated.

In the hypothetical case, the Seattle-San Diego subregion would be covered by a spot beam containing 26 channels. Six of these would be for internal zone use, 11 would be dedicated to the eastern subregion of the U.S., 5 would be for the midwest subregion including Houston and St. Louis, and 4 would be for the Denver-Phoenix mountain subregion. With each circuit required to be duplex (two-way), the 84 channels would provide 42 duplex circuits. This scheme would be operable but would not afford the flexibility needed to reroute the circuits in order to equalize traffic loads. It is not efficient in another aspect; dedication of channels on a peak load basis would necessitate operation of transponders and associated hardware at under capacity for most of the schedule. Time differences in zonal CONUS produce obvious reversals of load between zones. Due to these considerations, a form of time division multiple access is favored that performs multiplexing at the channel level.

Channel Reallocation Concept

For the same scenario as the FDMA example described, a scheme to provide TDMA on strictly a channel basis is shown in Figure 4.2-7. Referring to the figure, 84 channels are available to each of four spot beams covering CONUS. These beams could be provided in another region, as required. The beams are the result of radiation patterns from four typical parabolic antennas used for reception/transmission. Each of the antenna reception feedlines are routed to a frequency demultiplexer and a set of wideband amplifiers for 12 channels each by means of hybrid couplers.

The K_{HI}-band is 3.5 GHz in bandwidth. This bandwidth affords division into seven bands 500 MHz wide. Each 500 MHz band can accommodate 12 channels occupying 40 MHz each. As shown in the figure, each channel is amplified to an appropriate power level and fed to a four-pole RF switch. The four-pole RF switches for each channel are grouped in an 84 by 4 switching matrix. The outputs of the matrix are routed to a frequency multiplexer that produces a 500 MHz bandwidth subgroup. The subgroups are then combined for each output at the appropriate antenna transmission feed for the four beams of the down-link as shown in Figure 4.2-8.

Switching of the matrix changes distribution of the channels. This is considered a basic scheme that can operate at various switching rates under ground control via the TT&C of the platform. The command system function of the TT&C, as described elsewhere in this report, can handle command rates as high as 64 bits per second for control of the matrix. A central switching control provides the timing required to the degree of precision necessary for

C3

K_{HI} BAND 84 CHANNEL CAPACITY

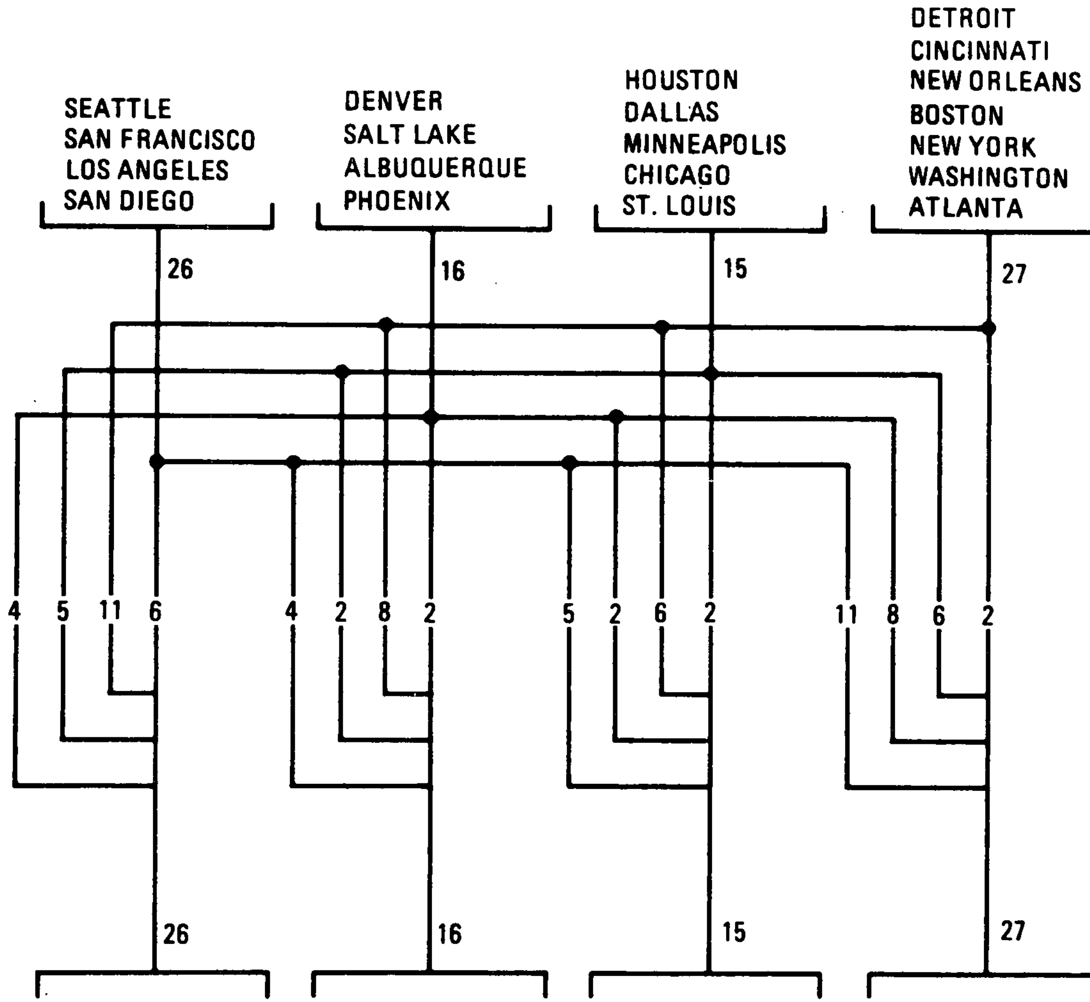


Figure 4.2-6. Fixed Channel Allocation Concept

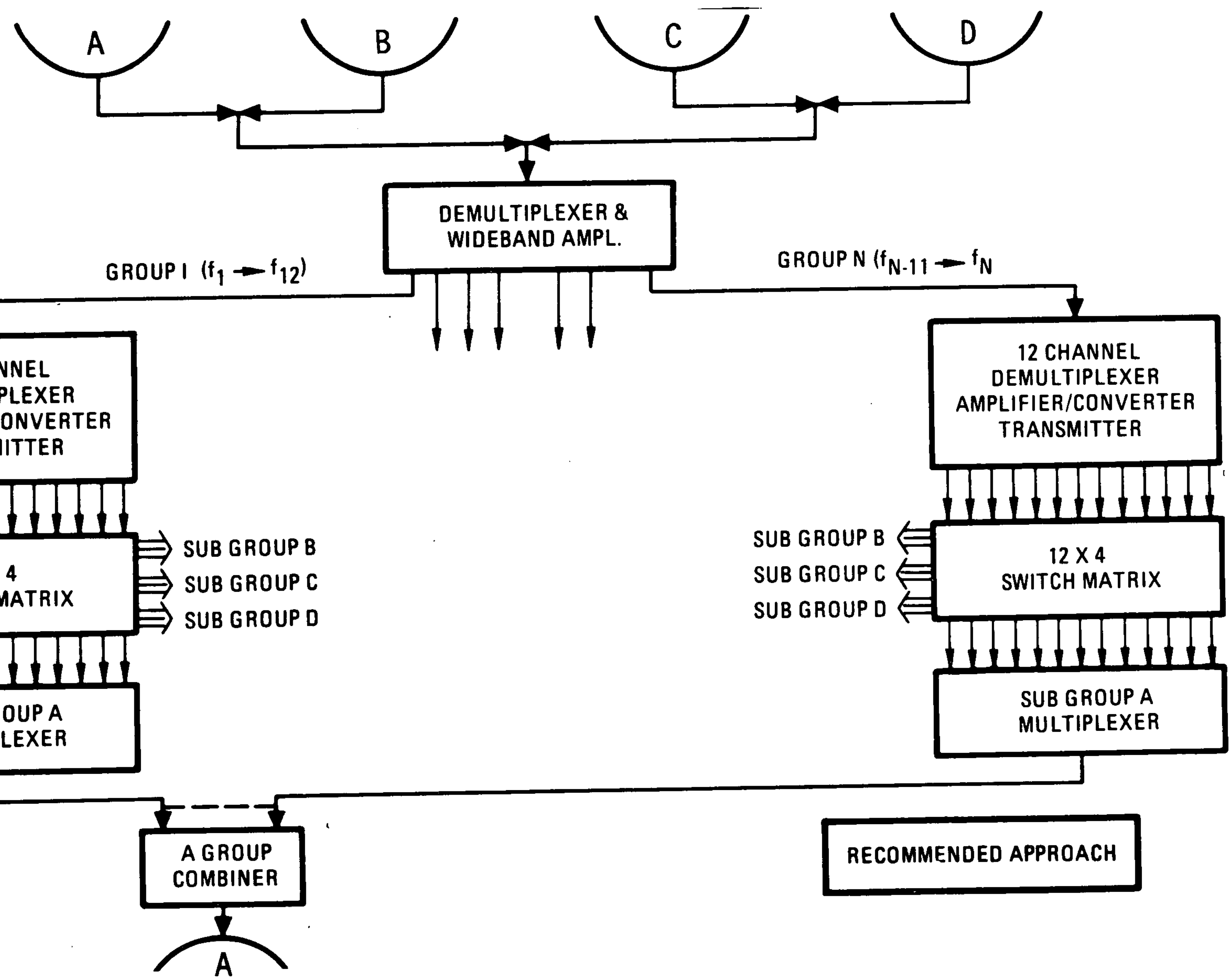


Figure 4.2-7. Channel Assignment Functional Block Diagram

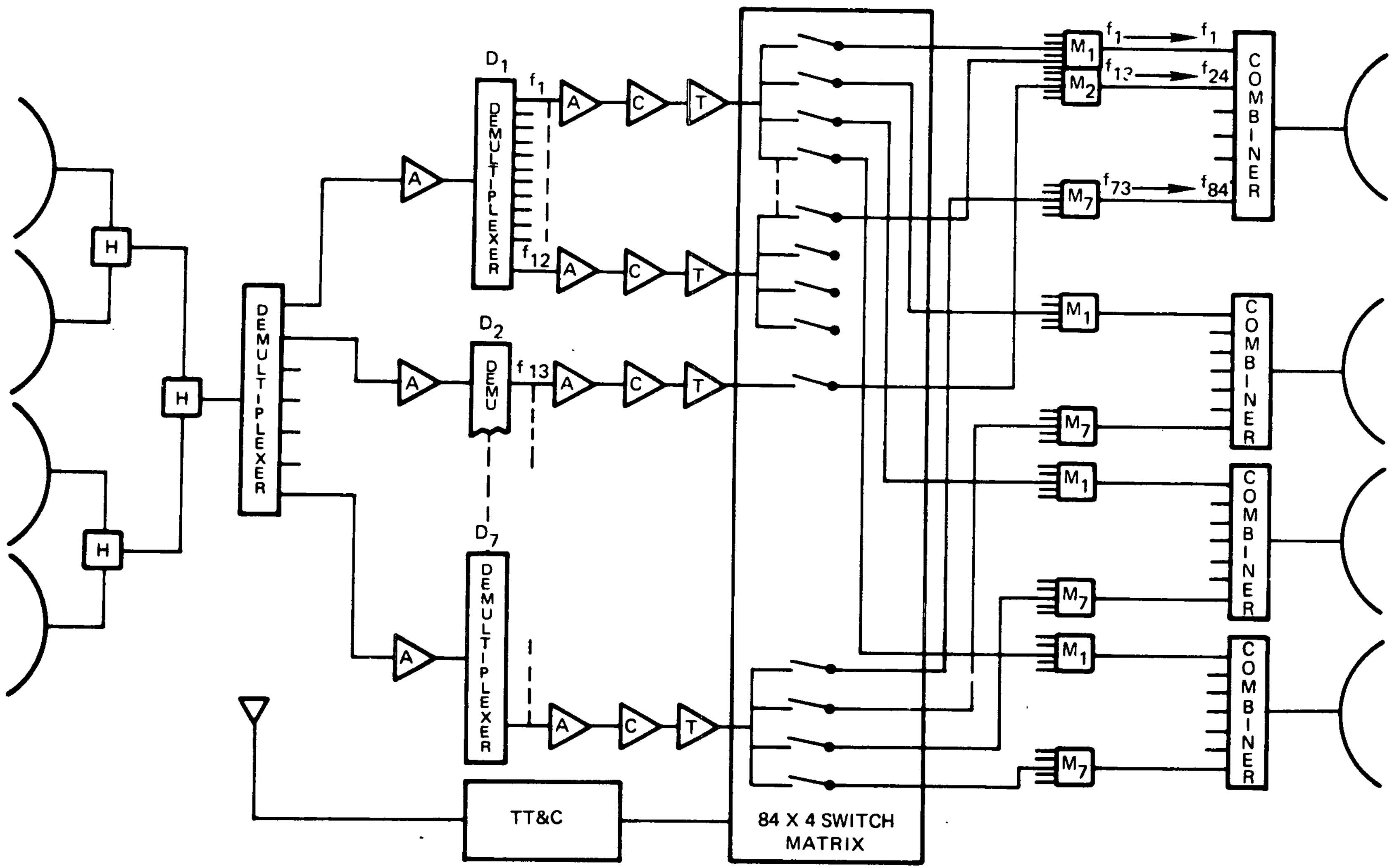


Figure 4.2-8. Channel Assignment Functional Schematic

multiple access. This can be designed for either a low level of scheduling sophistication or, ultimately, for rapid burst transmission as visualized in the literature for advanced heavy traffic systems.

Control of the switching is postulated as a ground function via the TT&C, but eventually could become autonomous on the platform. The literature describes autonomous platform systems that require users to acquire synchronism from the satellite clock signals emitted in the downlink. These schemes would be applicable to the traffic loads envisioned for second- or third-generation platforms. For the initial concept, ground control of switching places the majority of control complexity on the ground with savings in weight and power on the platform due to minimizing of onboard memory and computation.

MODULAR PACKAGING CONCEPTS

The dimensions of the modules comprising each platform communication system are based on projected hardware characteristics in the 1980 time period. Advances are also assumed in the technology associated with microelectronics, striplines, acoustic wave devices, and solid-state RF switching elements.

C-Band Hardware Elements

Synthesis of the C-band system for a platform is based on existing hardware designs extended to include anticipated reductions in size realizable in the late 1970's. For instance, buildup of switch matrices evolves from a one by four switch assembly as shown in Figure 4.2-9. The switching elements are conventional pin diodes in SP4T configuration. The sizing parameters are essentially a function of the standard connector dimensions and necessary physical spacing for access and mounting. The unit uses typical SMA connectors for the RF terminals of the switch, and a typical electrical multi-pin connector for the pin diode driver inputs. This module can be stacked in quad assemblies to provide 84 inputs and 84 by 4 outputs (Figure 4.2-10). Total package size is 21 inches by 10 inches by 8 inches. For a C-band system consisting of 24 channels and 8 antennas, which would be usable in all four regions, a matrix switch of 24 by 8 is required. The packaging dimensions for nested assemblies that permit replacement as a unit are shown in Figure 4.2-11. The platform interfaces are shown in the "A" insert drawing. A 12-channel to 4-antenna assembly would appear as in the "B" insert drawing. These package designs are predicated on use of miniaturized filter designs. They do not reflect waveguide-type filters currently in use.

K-Band Hardware Elements

A modular communication package for the K-band frequencies is shown in Figure 4.2-12. The figure gives the dimensions for a system capable of operation in either the K_{LO}- or K_{HI}-bands. In the example depicted, an 84 by 4 matrix switch is used. Platform interface connections consist of the four antenna outputs, a single RF input, and control and dc power inputs through a common connector. As in the case of C-band hardware, the connections are all visualized on the same side for unit module replacement.

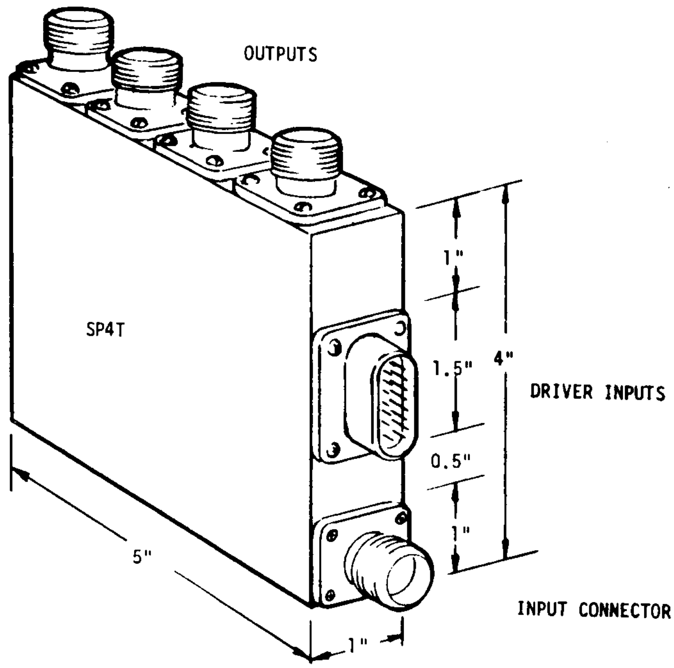


Figure 4.2-9. Single 1 x 4 Switch Module Dimensions

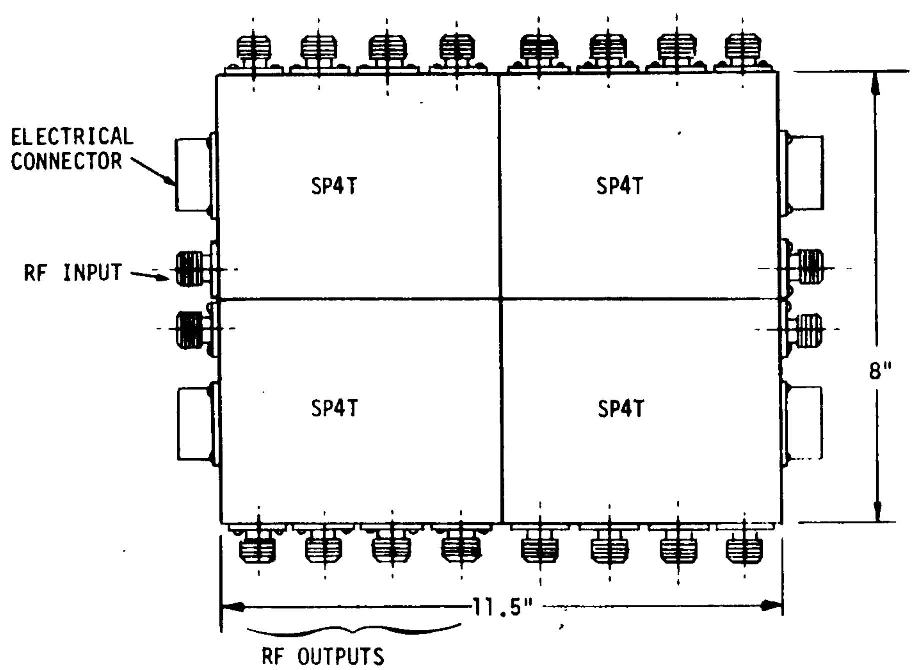


Figure 4.2-10. Quad Assembly of 84 x 4 Switching Matrix

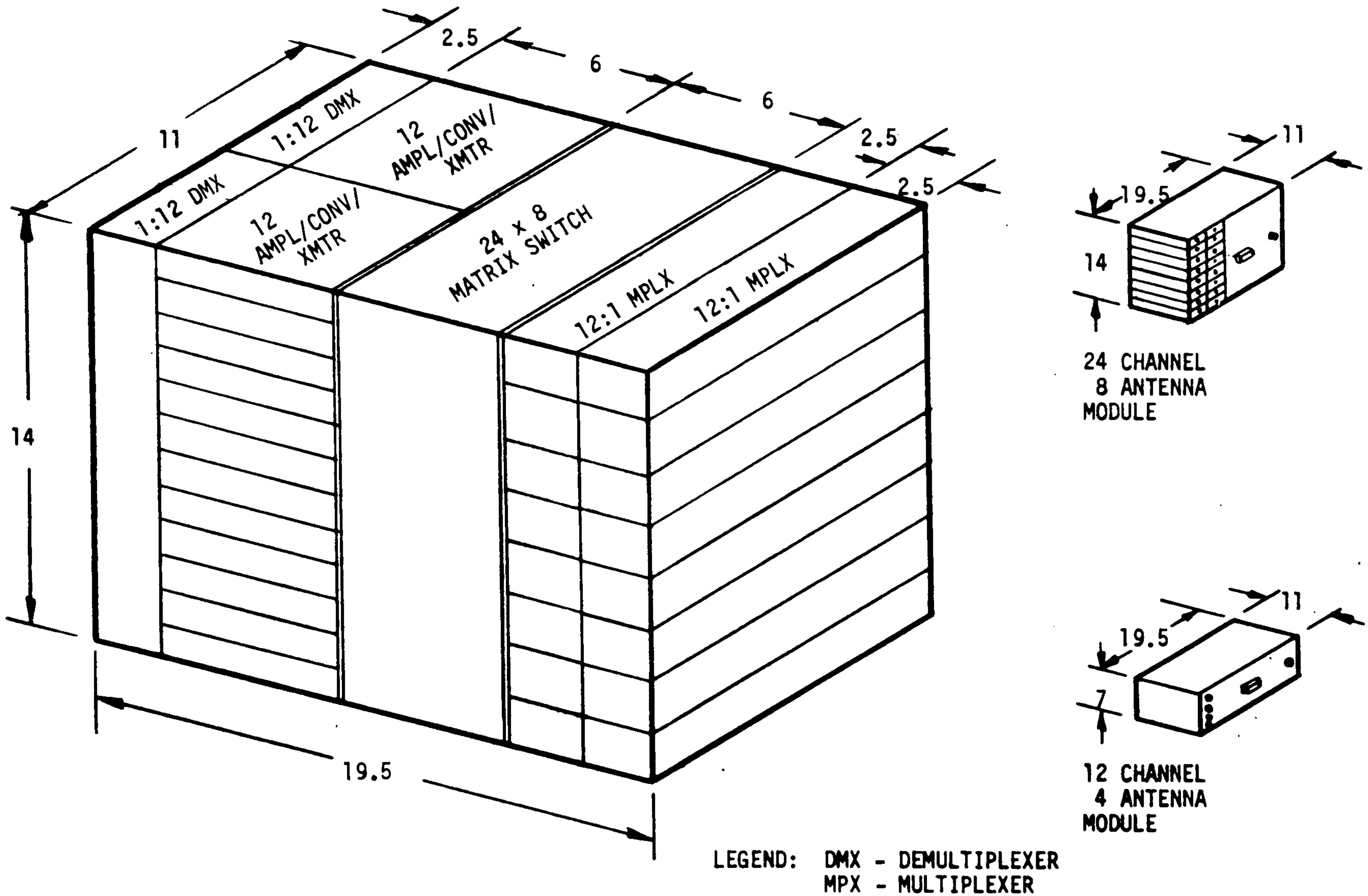


Figure 4.2-11. C-Band Transponder Module

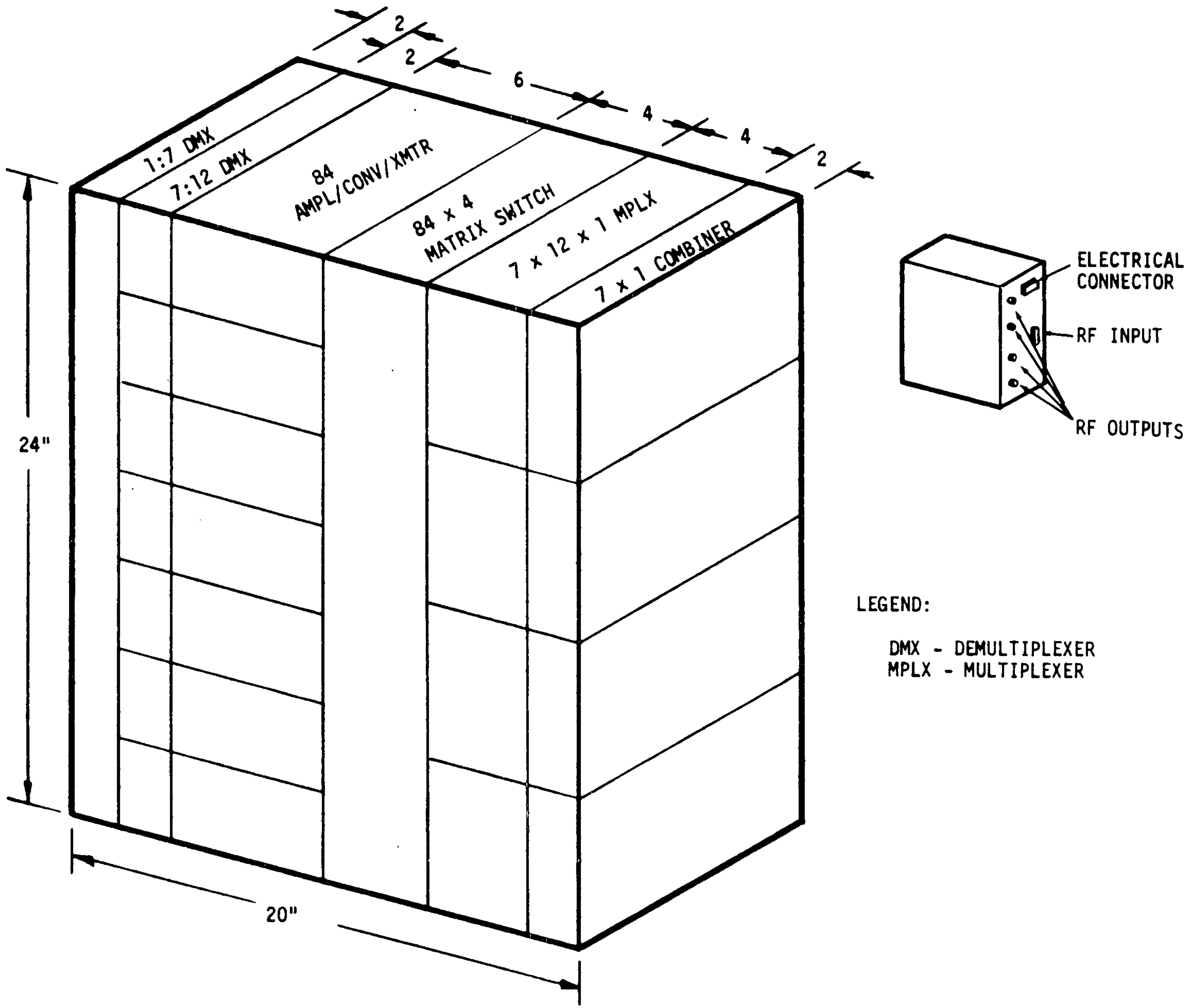


Figure 4.2-12. K-Band Transponder Module

Navigation and Traffic Control Hardware Elements

For the navigation and traffic control functions of the data relay platform, an L-band phased array with dimensions as shown in the outline drawing of Figure 4.2-13 is visualized. The electronics package containing the L-band transponder and associated equipment is postulated as a rectangular box 12 inches high, 15 inches wide, and 20 inches in depth, which is readily accommodated in the standard sized module. The weight of this package is estimated at 105 pounds. The replaceable L-band module is shown in Figure 4.2-14.

The transmit/receive functions of the transponder could be incorporated into the array assembly to comprise an integrated phased array. As in the case of the state-of-the-art ground systems, the distribution of multiple elements to drive the array and others to provide reception/preamplification affords a gradual degradation characteristic in lieu of sudden failure of the unit transponder.

However, the integral packaging concept would require changeout of the entire array periodically. The preferred approach is to package the electronics separately to facilitate on-orbit servicing. The totally passive characteristics of the array in this approach makes it extremely unlikely that the array will require servicing during the platform's mission life.

DATA RELAY PLATFORM DESCRIPTIONS

Layout drawings of the types of data relay platforms for each region of the world are presented in Figures 4.2-15, 4.2-16, 4.2-17, and 4.2-18. Each platform consists of a common support module (defined in Section 3.0), a mission equipment module, and the appropriate antenna configuration. Platform orientation during operation is indicated. The platform configurations for shuttle delivery (stowed antennas) are also shown on the layout drawings.

It is apparent that a mission equipment module or ring that can accommodate 12 replaceable modules is not necessary for any of the data relay platforms that satisfy the requirements of the baseline traffic model. Alternate configurations were evaluated, but none provided the adaptability for both manned and unmanned servicing or the commonality of equipment.

In previous sections of this report it was pointed out that one of the primary reasons for the inside changeout concept was to facilitate the progression of servicing modes from auto/remote to shirtsleeve. Auto/remote inside changeout requires a relatively large clear area in the center of the toroid for maneuvering of the manipulator, its extension arms, and an attached module. Also in manned servicing mode a reasonable work space must be provided within the platform for manual handling of modules. If the data relay platforms consisted of only the common support module ring and the mission equipment was essentially cantilevered into the center of that ring from the back of the L-band phased array structure, adequate space would not be available for any of the servicing modes.

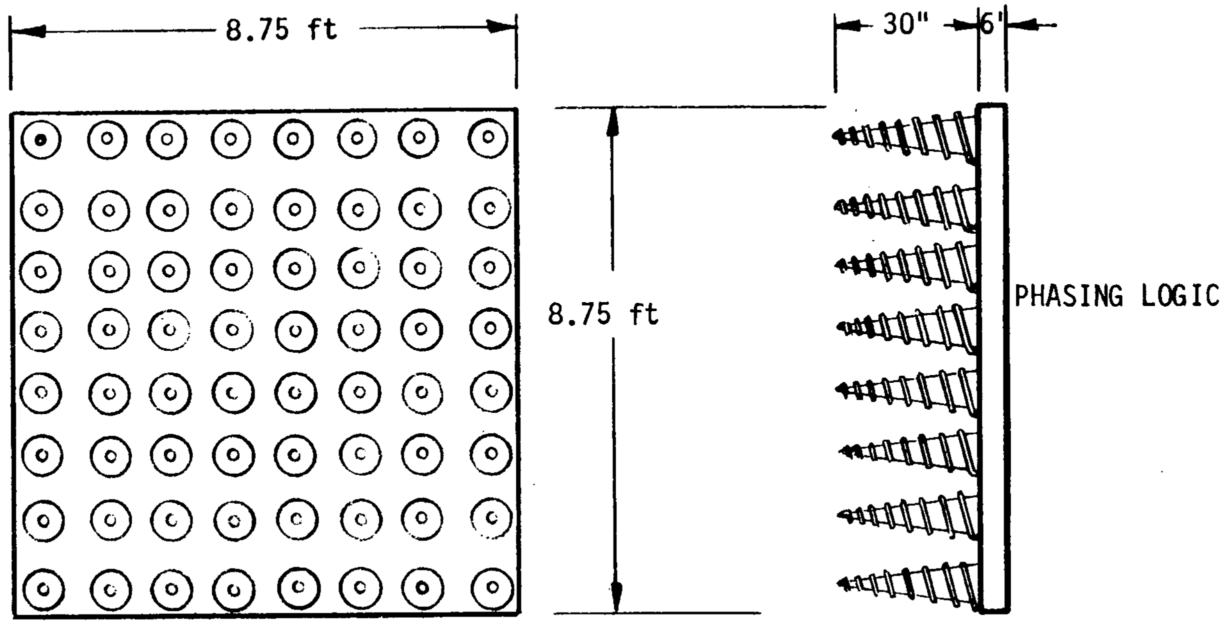


Figure 4.2-13. L-Band Electronically Steered Phased Array

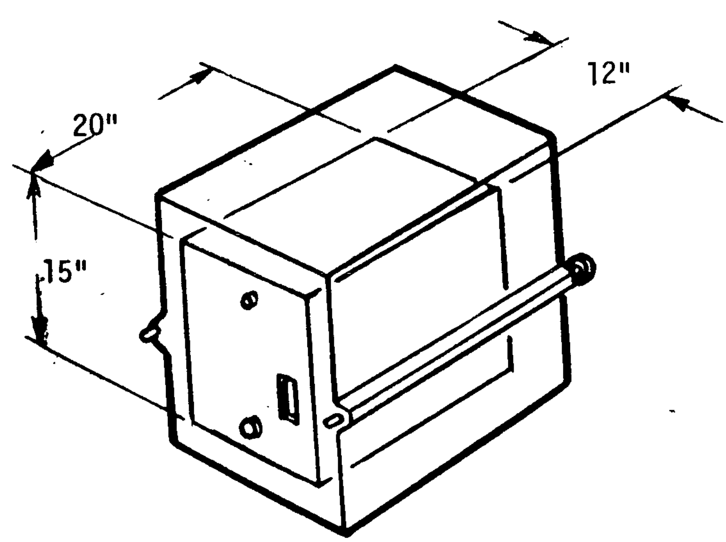
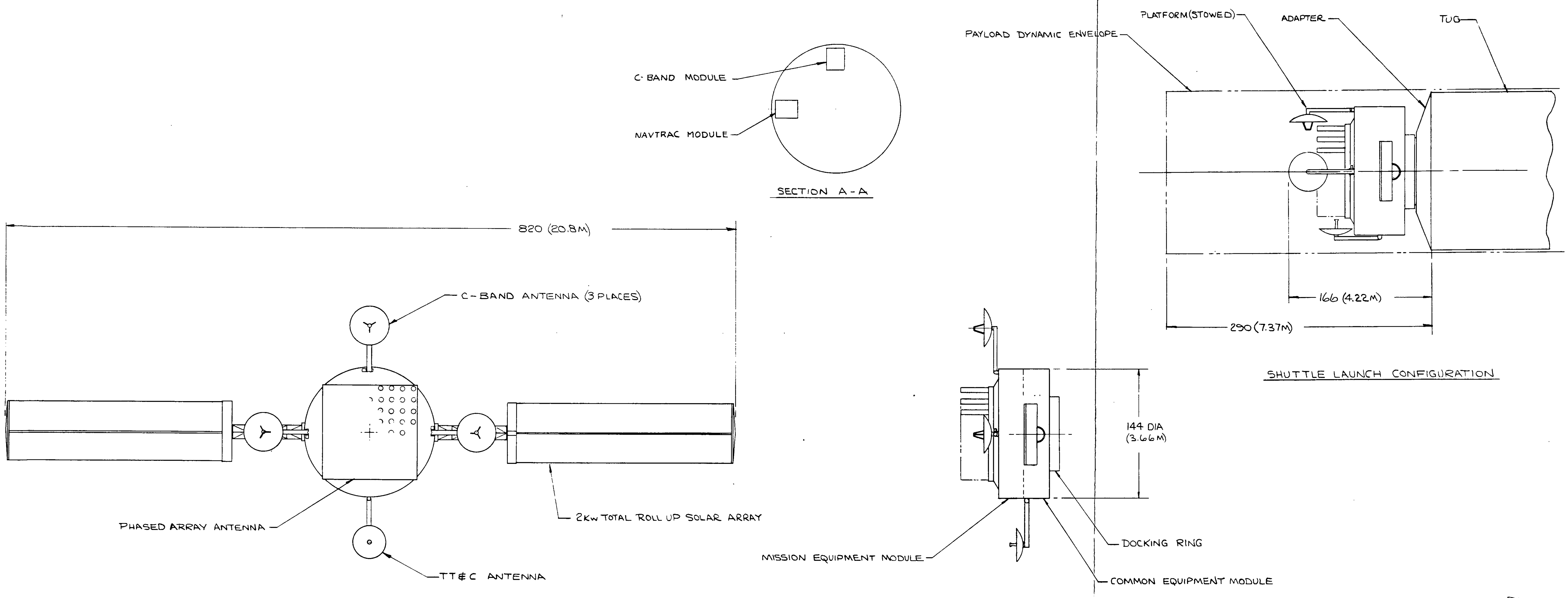


Figure 4.2-14. L-Band Transponder Module



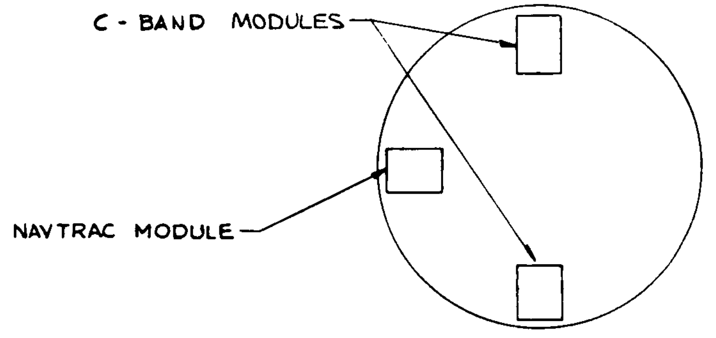
FOLDOUT FRAME 3

Figure 4.2-15 SD 73-SA-0036-5 4-41, 4-42

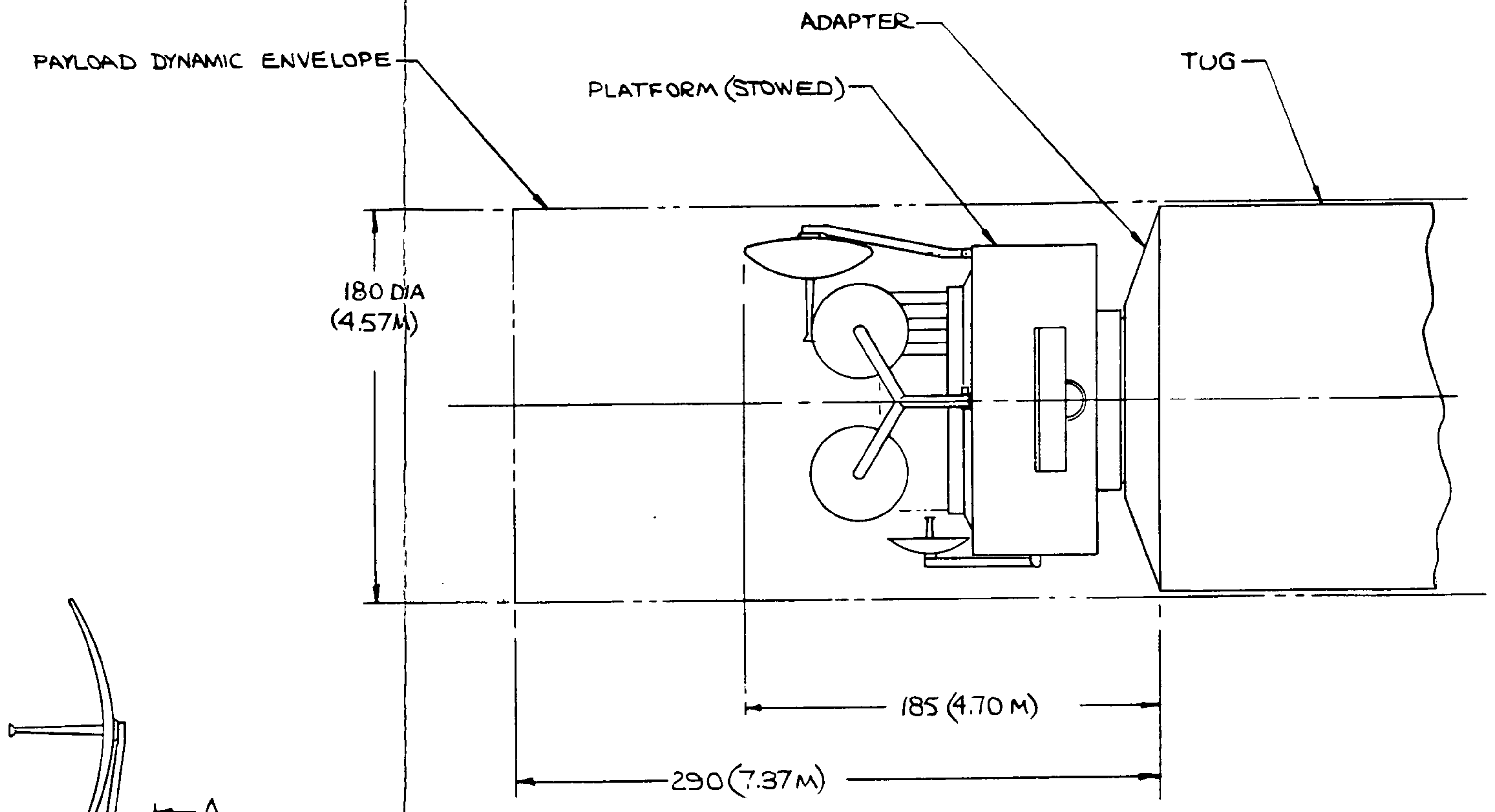
SCALE 1/40	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
REGION I DATA RELAY PLATFORM GEOSYNCHRONOUS PLATFORM DEFINITION STUDY		2339-12

FOLDOUT FRAME 1

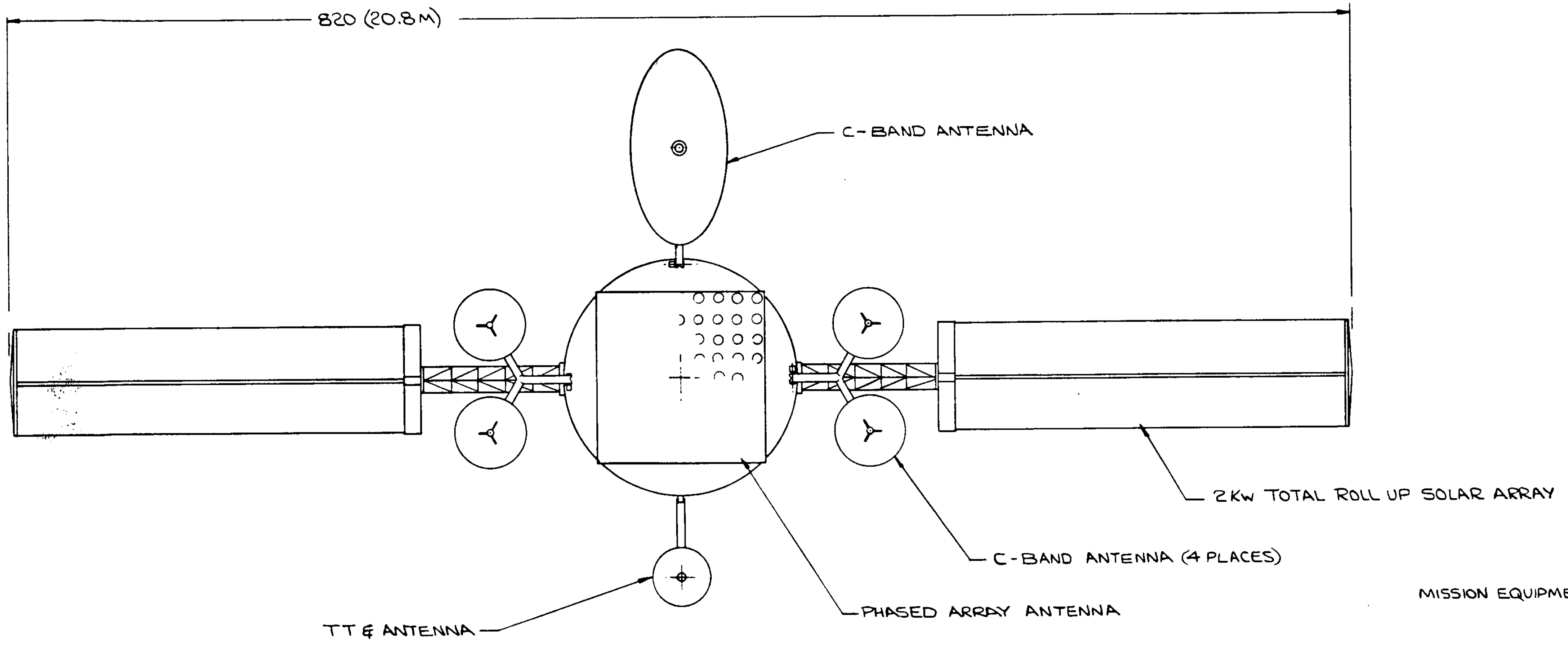
FOLDOUT FRAME 2



SECTION A-A



SHUTTLE LAUNCH CONFIGURATION



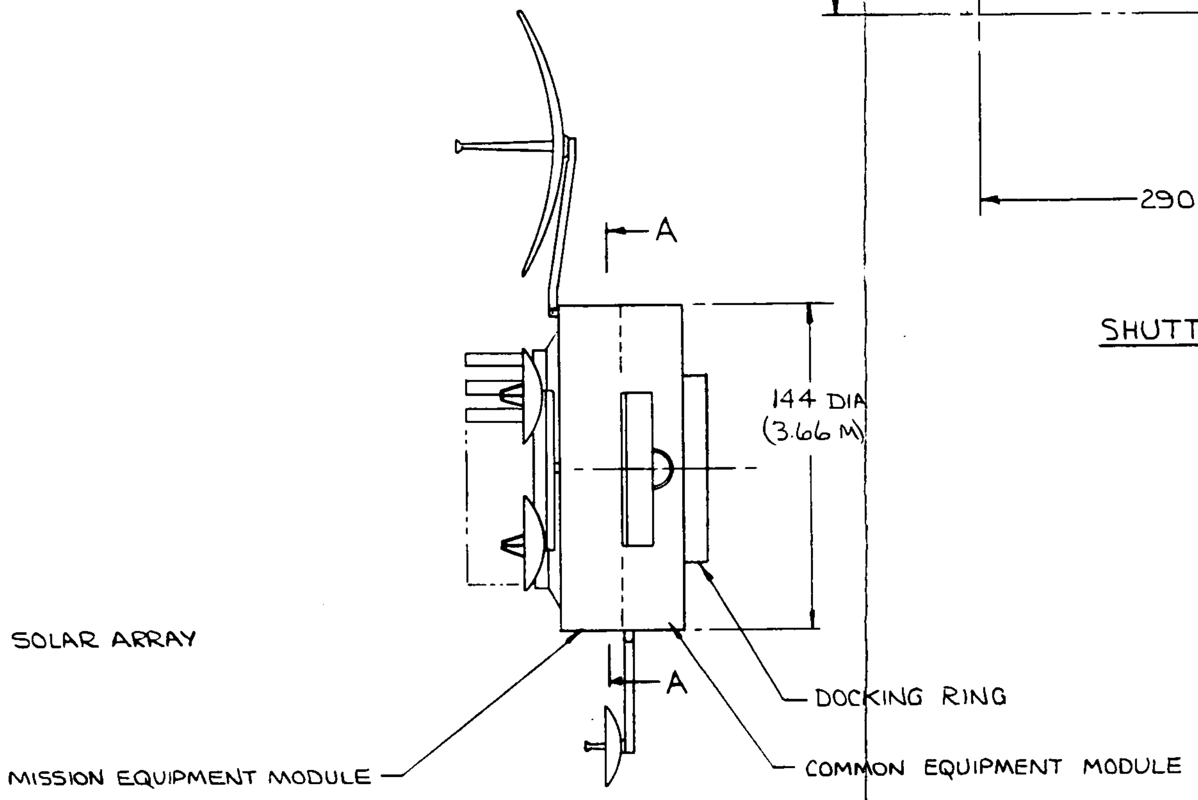
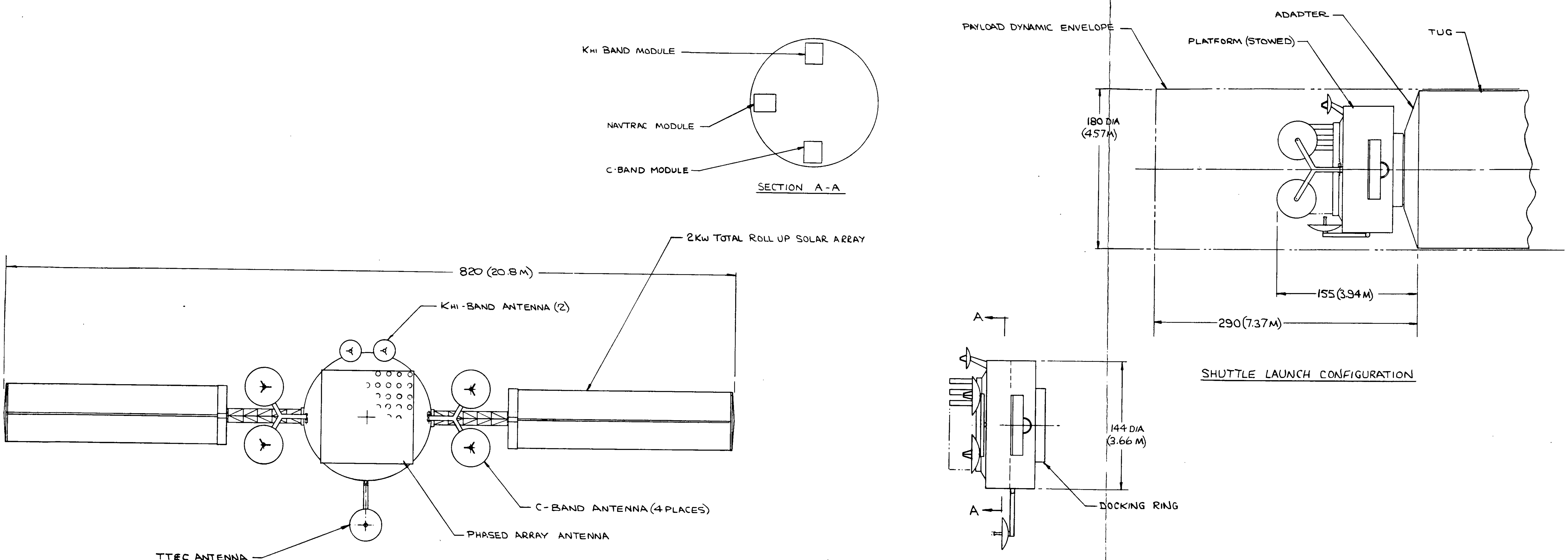


Figure 4.2-16 SD 73-SA-0036-5

B
4-43, 4-44

SCALE 1/40		SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
REGION II DATA RELAY PLATFORM GEOSYNCHRONOUS PLATFORM DEFINITION STUDY		2339-13



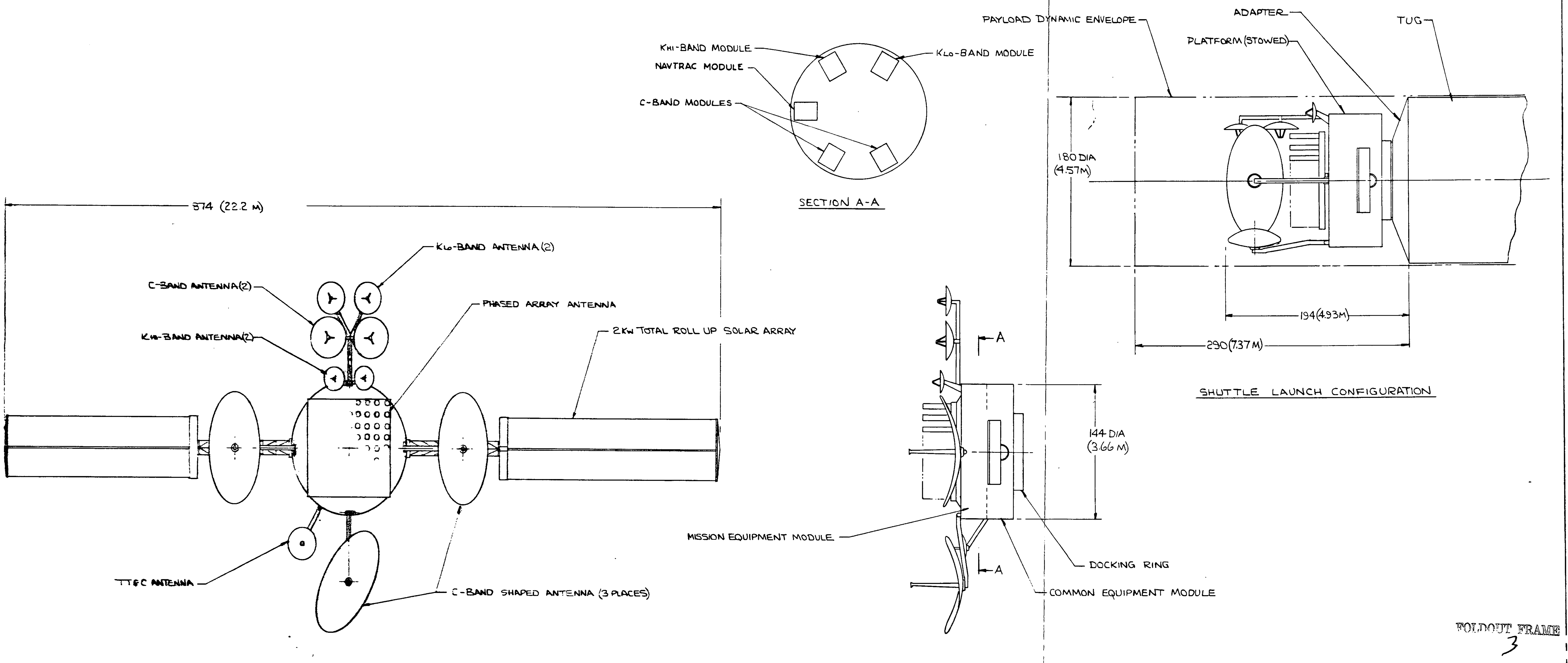
FOLDOUT FRAME 3

Figure 4.2-17 SD 73-SA-0036-5 4-45, 4-46

1/40		SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION	
REGION III DATA RELAY PLATFORM GEOSYNCHRONOUS PLATFORM DEFINITION STUDY			2339-14

FOLDOUT FRAME 1

FOLDOUT FRAME 2



FOLDOUT FRAME 3

Figure 4.2-18 SD 73-SA-0036-5 4-47, 4-48

SCALE 1/40	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
REGION IV DATA RELAY PLATFORM		2339-15
GEOSYNCHRONOUS PLATFORM DEFINITION STUDY		

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Housing the data relay platform mission equipment in a structure that does not interfere with the free space of the common support module did not provide weight or volume savings that warranted the additional design. The L-band array requires some type of structure that has approximately a 12-foot diagonal. Adequate room for replacement of the mission equipment module must be provided. The mission equipment structure must be pressurizable for the shirtsleeve servicing mode. Thus, some type of a cylinder is required that will support the L-band array and is connected to the common support module. Admittedly, an optimized design would not be a 12-foot ring. However, the differences would not be significant and a separate design and development is not warranted. Use of the standard common support module ring is the recommended approach for installation of data relay platform mission equipment.

A tabulation of the weights and power requirements of each type of platform is presented in Tables 4.2-1, 4.2-2, 4.2-3, and 4.2-4.

Table 4.2-1. Region I Data Relay Platform Summary

	Item	Weight (lb)		Power (watts)	
		Subtotal	Total	Subtotal	Total
Operations Equipment	Structures		465		--
	Primary	100			
	Secondary	300			
	Thermal protection	65			
	Mission Equipment		305		600
	C-band transponder	130		300	
	C-band antennas (3)	30		--	
	L-band transponder	105		300	
	L-band antenna	40			
Common Support Module	Structure		500		--
	Subsystems		1489		218
	Totals		2759		818

Table 4.2-2. Region II Data Relay Platform Summary

	Item	Weight (lb)		Power (watts)	
		Subtotal	Total	Subtotal	Total
Operations Equipment	Structures		530		--
	Primary	100			
	Secondary	300			
	Thermal protection	130			
	Mission Equipment		467		1200
	C-band transponders (2)	260		600	
	C-band antennas (5)	62			
	L-band transponders	105		600	
	L-band antenna	40			
Common Support Module	Structure		500		--
	Subsystems		1489		218
	Totals		2986		1418

Table 4.2-3. Region III Data Relay Platform Summary

	Item	Weight (lb)		Power (watts)	
		Subtotal	Total	Subtotal	Total
Operations Equipment	Structure		530		--
	Primary	100			
	Secondary	300			
	Thermal protection	130			
	Mission Equipment		755		1080
	C-band transponder	130		300	
	C-band antennas (4)	40			
	K_{HI}-band transponder	420		480	
	K_{HI}-band antennas (2)	20			
	L-band transponder	105		300	
	L-band antenna	40			
Common Support Module	Structure		500		
	Subsystems		1489		218
	Totals		3274		1298

Table 4.2-4. Region IV Data Relay Satellite Summary

	Item	Weight (lb)		Power (watts)	
		Subtotal	Total	Subtotal	Total
Operations Equipment	Structure		595		--
	Primary	100			
	Secondary	300			
	Thermal protection	195			
	Mission Equipment		1251		1505
	C-band transponders (2)	260		600	
	C-band antennas (5)	86			
	K_{LO}-band transponder	300		125	
	K_{LO}-band antennas (2)	20			
	K_{HI}-band transponder	420		480	
	K_{HI}-band antennas (2)	20			
	L-band transponder	105		300	
	L-band antenna	40			
Common Support Module	Structure		500		--
	Subsystems		1489		218
	Total		3835		1723

IMPLEMENTATION PLAN

The baseline traffic model extended from 1973 through 1990. During the 1980's, the shuttle era, several of the expendable satellites listed in the traffic model are replacements for pre-1980 satellites. Others are expansion of data relay capacity. The criteria for introduction of platforms into the geosynchronous program was to schedule the launch of a platform when either a replacement satellite or an expansion satellite was required after 1979.

The satellite launch schedule and platform schedule for each of the four regions are presented in Figures 4.2-19, 4.2-20, 4.2-21, and 4.2-22. The satellites that are replaced by platforms are indicated. The legend used in each figure is as follows:

▲ = Expendable satellite launch

■ = Platform launch

○ = Platform capability in excess of traffic model requirements

= Platform equivalent of expendable satellites

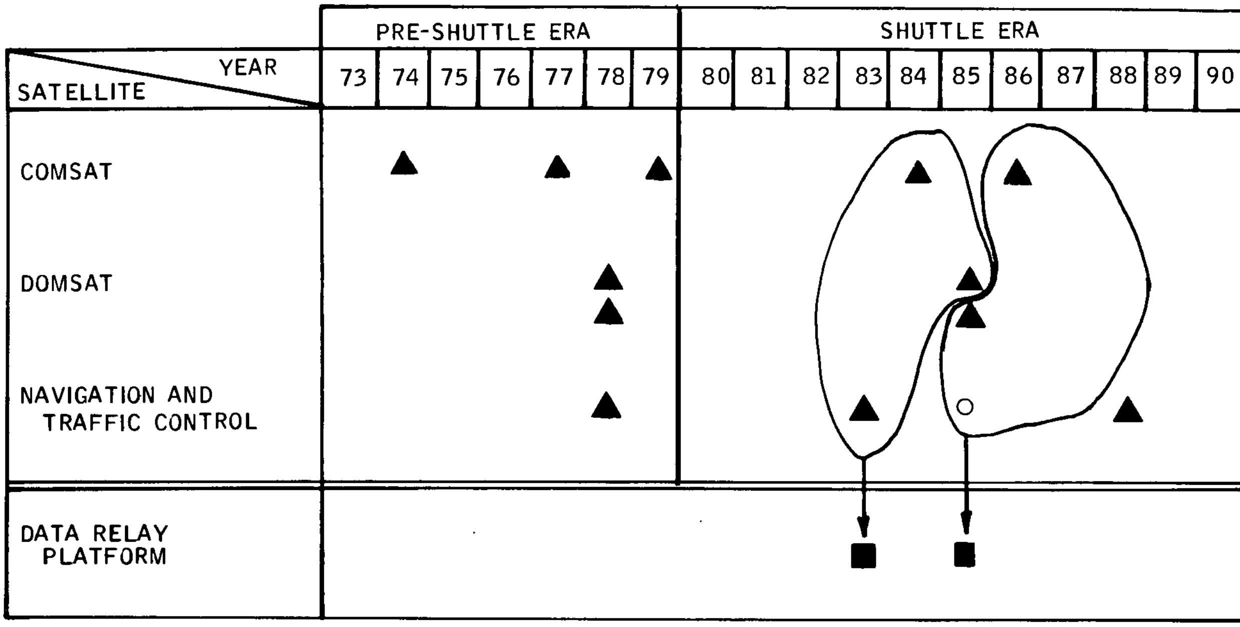


Figure 4.2-19. Region I Data Relay Platform Implementation Schedule

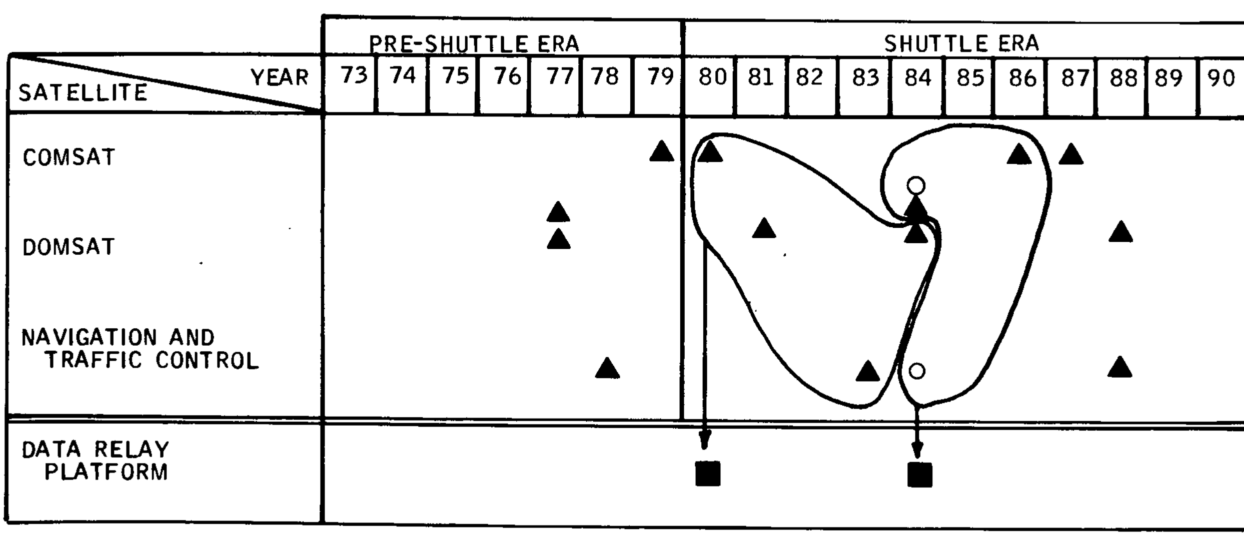


Figure 4.2-20. Region II Data Relay Platform Implementation Schedule

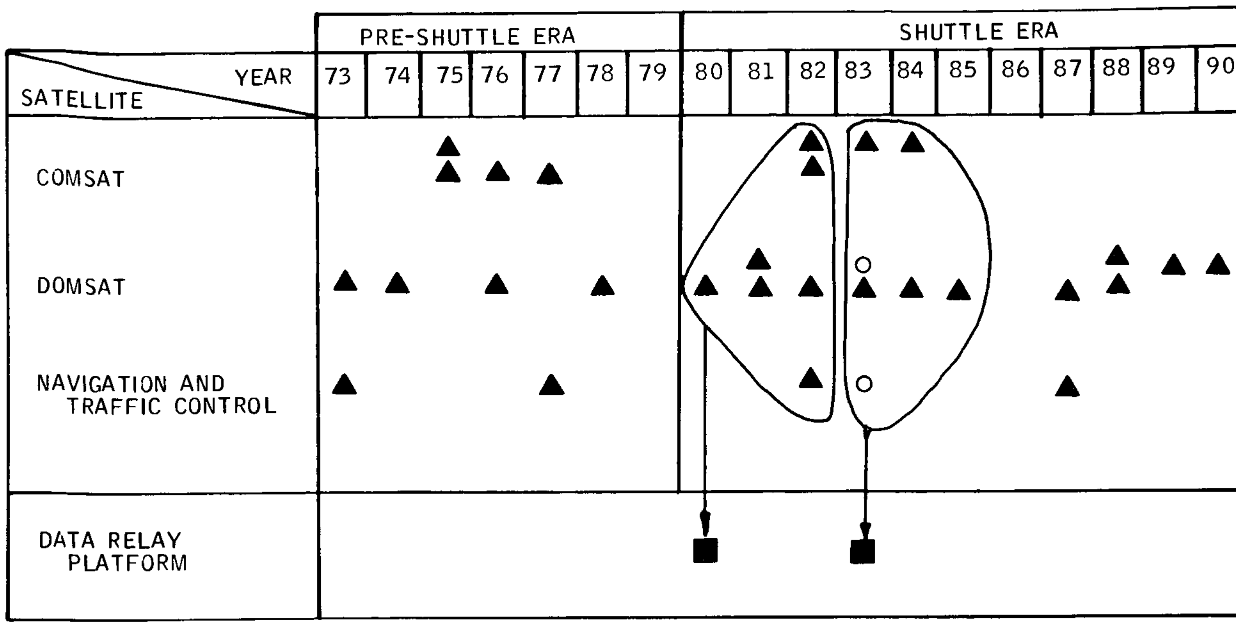


Figure 4.2-21. Region III Data Relay Platform Implementation Schedule

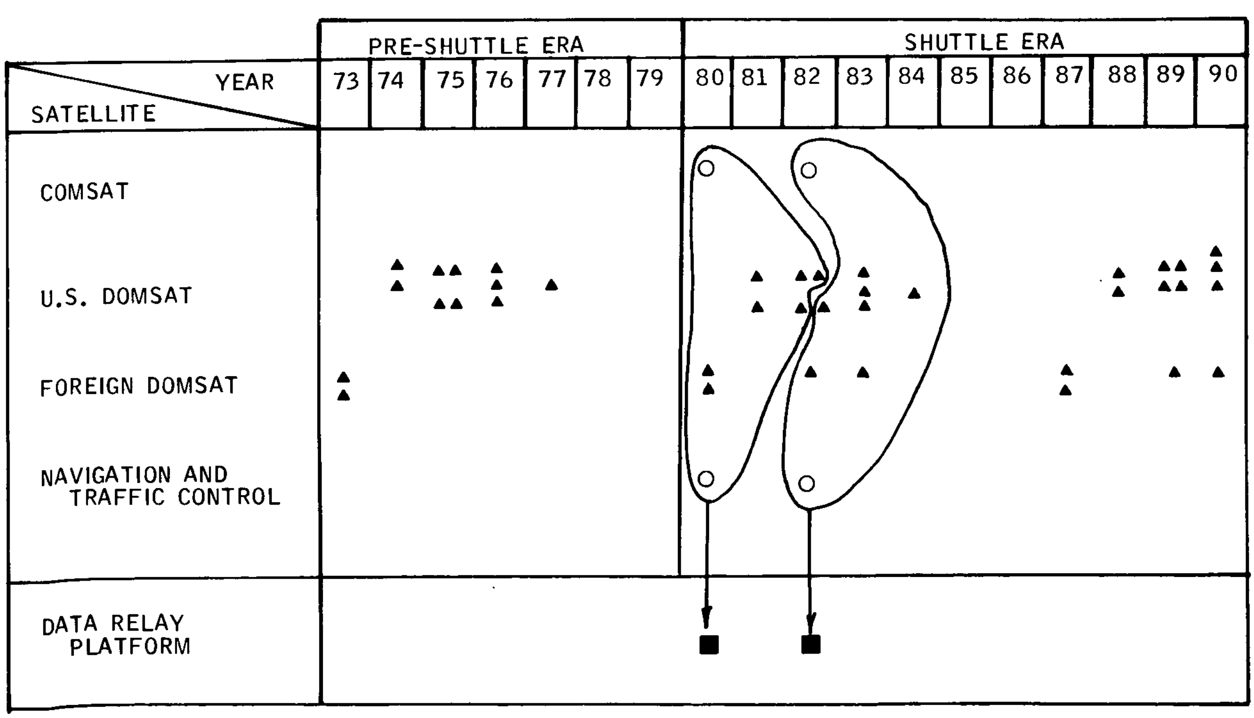


Figure 4.2-22. Region IV Data Relay Platform Implementation Schedule

Note that in all regions, expendable satellites in the late 1980's are not replaced by platforms. This is indicative of the on-orbit servicing concept included in the platform program. Where satellite replacement is scheduled, platform refurbishment would occur.

Both Comsat and navigation and traffic control expendable satellites were added to the Region IV schedule associated with the baseline traffic model in order to reflect the capability of the platform.

4.3 TDRS PLATFORM SYNTHESIS FOR BASELINE TRAFFIC MODEL

Detailed design trades are currently being conducted in Phase B NASA contractual studies. In this section a synopsis of one of the candidate concepts which is representative of a TDRS platform is presented. The performance characteristics are delineated. Design concepts including functional block diagrams and equipment descriptions are identified. Packaging of the mission equipment for adaptation to the geosynchronous platform is derived and a layout of the integrated TDRS platform is presented.

PERFORMANCE CHARACTERISTICS

The primary functions of TDRS are: relay of low, medium, and high data rates to and from low earth orbiting satellites and to and from a single ground terminal. In addition, TDRS is to provide tracking data and priority command service. The gross system characteristics of each function are described below.

Low Data Rate Service

The low data rate (LDR) service is provided by UHF command links with two TDRS steered beams and a high-gain antenna. System EIRP at UHF is +30 dBw per beam minimum. The telemetry link uses an adaptive ground implemented phased array (AGIPA). In prior studies on the TDRS concept and related services, the RFI problem and multipath phenomena have been the most important technical problems. Studies by Magnavox and AIL on pseudonoise (PN) modulation techniques and AGIPA antennas have suggested means to solve these problems. For LDR service, the use of AGIPA concepts provides 20 independent beams with one beam for each user. User spacecraft are discriminated by means of unique PN codes in the return link. The AGIPA concept provides space and polarity filtering of RFI for optimization of the signal-to-interference ratio. Features of the design for LDR services are listed in Table 4.3-1. The AGIPA approach is shown schematically in Figure 4.3-1.

Table 4.3-1. Design Features for LDR

Forward Link	
Frequency band	400.5 to 401.5 MHz
Polarization	Circular
EIRP at 31 deg FOV	Steered beam: 30 dBw data; 36 dBw voice at 25-percent duty cycle; FFOV, 24 dBw
Return Link	
Frequency band	136 to 138 MHz
Polarization	Linear/2 planes
G/T (deg K)	-14.4 dB (AGIPA); -18.8 dB (FFOV)
Antenna	Backfire, four-element array
Transceiver Type	Translator; nominal temperature, 800 K

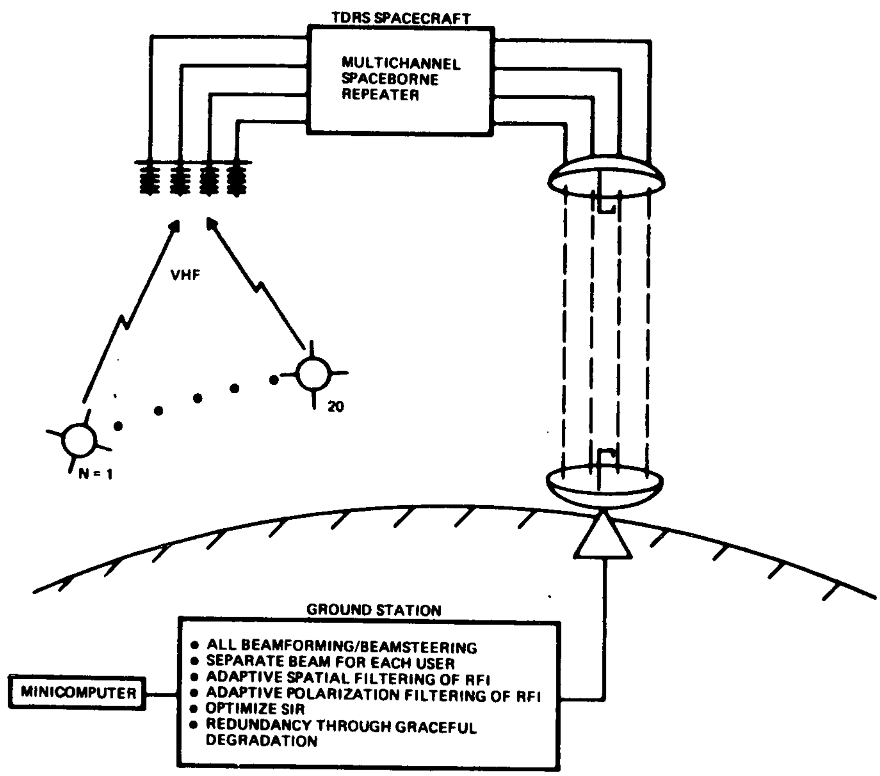


Figure 4.3-1. AGIPA Approach for LDR Return Link

Medium Data Rate (MDR) Service

MDR user service comprises a dual frequency concept; an S-band link supports near-term users, and a Ku-band link supports future high data rate (HDR) users. Two MDR users can be supported simultaneously via two 12.5-foot reflector antennas on the platform. The MDR service provides support to manned spacecraft users (space shuttle). Antenna tracking is open loop at S-band and auto track at Ku-band. MDR service features are listed in Table 4.3-2.

High Data Rate (HDR) Service

HDR user service includes data rates of up to 50 megabits per second. The design features for HDR service are listed in Table 4.3-3.

Table 4.3-2. Design Features of MDR Service

Forward Link	
Frequency	S/Ku-band
EIRP	S-Band: Data - 41 dBw Voice (25 percent duty cycle) - 47 dBw Ku-Band: 45.6 dBw
Polarization	Circular
Return Link	
Frequency	S/Ku-band
G/T	3.9/20.4 dB
Polarization	Circular
Antenna	Two S/Ku-bands; parabolic reflectors
Transponder Type	Translator

Table 4.3-3. Design Features of HDR Service

Forward Link	
Frequency	S-band or Ku-band
RF bandwidth	S-band: 32 MHz in a 75 MHz range Ku-band: 100 MHz
EIRP	S-band: 41 dBw unmanned; 47 dBw manned Ku-band: 23.6 dBw unmanned; 53.6 dBw manned
Redundancy	100 percent of active components
Return Link	
Frequency	S-band and Ku-band
RF bandwidth	S-band: 10 MHz in a 60 MHz range Ku-band: 150 MHz
G/T_S (deg K)	S-band: 10 dB; Ku-band: 25.9 dB
Redundancy	100 percent of active components
Backup Mode	
Frequency	S- or Ku-band
Mode	FDM

Tracking and Order Wire Functions

Based on conclusions reached in present studies by Rockwell on the TDRS, certain added design features are contemplated for more advanced configurations. These include more explicit identification of the TDRS tracking function and the provision for an order wire function. The TDRS tracking function is conceived to provide tracking and position location of TDRS spacecraft via two remote and the main TDRS ground stations. The order wire function provides capability for requests for priority access to TDRS medium data rate and/or high data rate transponder service.

DESIGN CONCEPTS

An integrated functional block diagram of the TDRS and also block diagrams of each major TDRS function are presented. The physical characteristics of each assembly (weight, power, and volume) are identified. An appropriate antenna configuration is also developed.

System Block Diagram

Elements that comprise the TDRS function as applied to the geosynchronous platform are shown in Figure 4.3-2, TDRS Telecommunication System Block Diagram. For LDR service, the system includes an 8-channel receiver operating with the AGIPA antenna and a 4-channel transmitter. For MDR/HDR service, two subsystems each are provided: an antenna diplexer feeding at S-/Ku-band receiver, and an S-/Ku-band dual-band transmitter. For the TDRSS to ground station link (TDRS/GS) function, input and output channel multiplexer and demultiplexer modules are included plus an FM modulator/transmitter and an HDR transmitter that feed a combiner. A ground link receiver is also provided that is diplexed with the output of the combiner to the common ground link antenna. Auxiliary functional modules are included in the system: (1) the TDRS tracking and order wire transponder, (2) the TT&C receiver and telemetry modules, and (3) the platform frequency source that supplies the various frequencies required.

MDR/HDR Transponder Design

A typical schematic for the MDR/HDR transponder functions is shown in Figure 4.3-3. Total weight of the S-/Ku-band subsystem, including the antenna, is estimated as follows:

	Pounds
Transponder weight	24.0
Antenna weight	39.4
Total	63.4

Power requirements for MDR/HDR modes are 18.2/66.0 watts at S-band, and 5.1/34.0 watts at Ku-band, respectively.

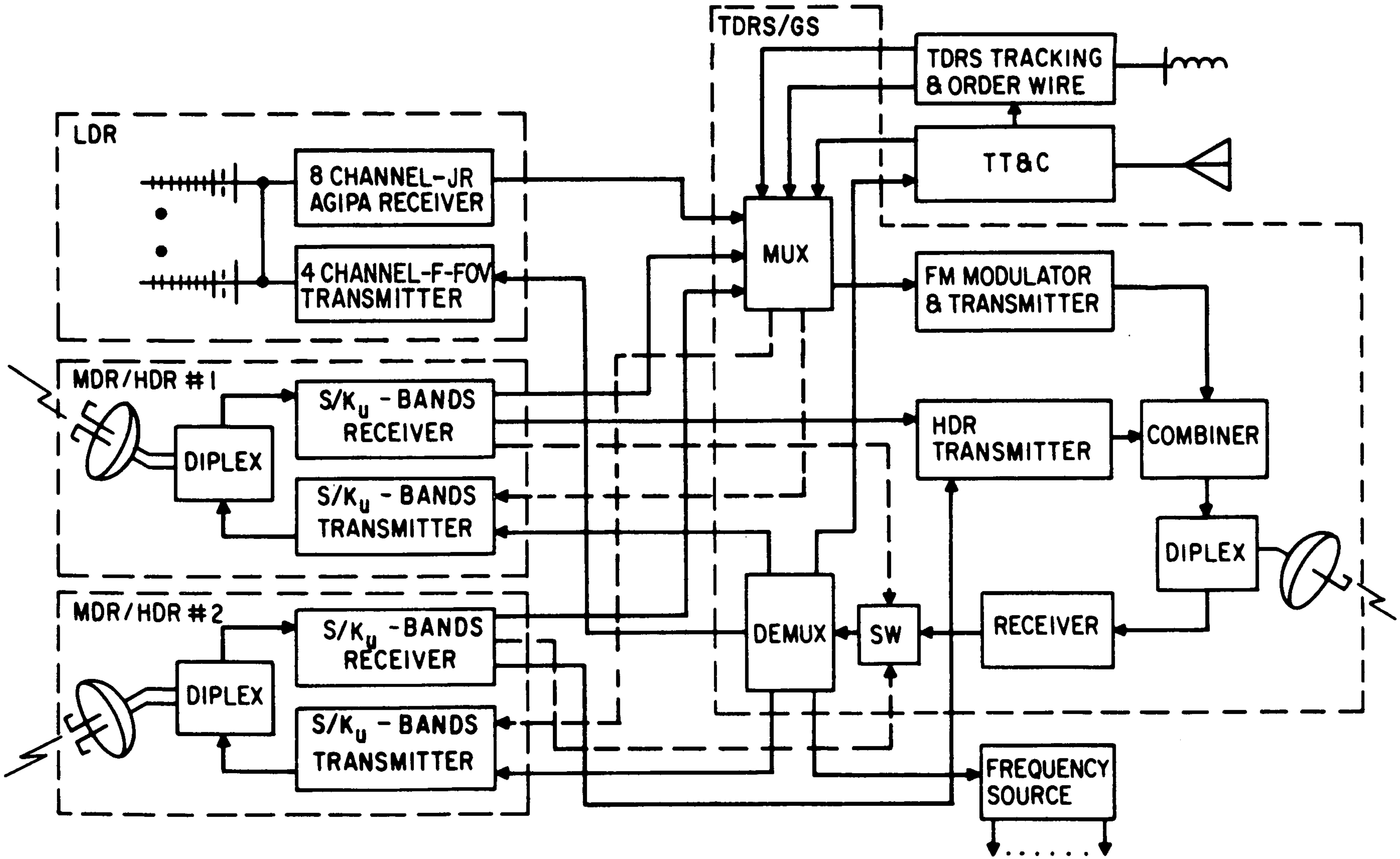
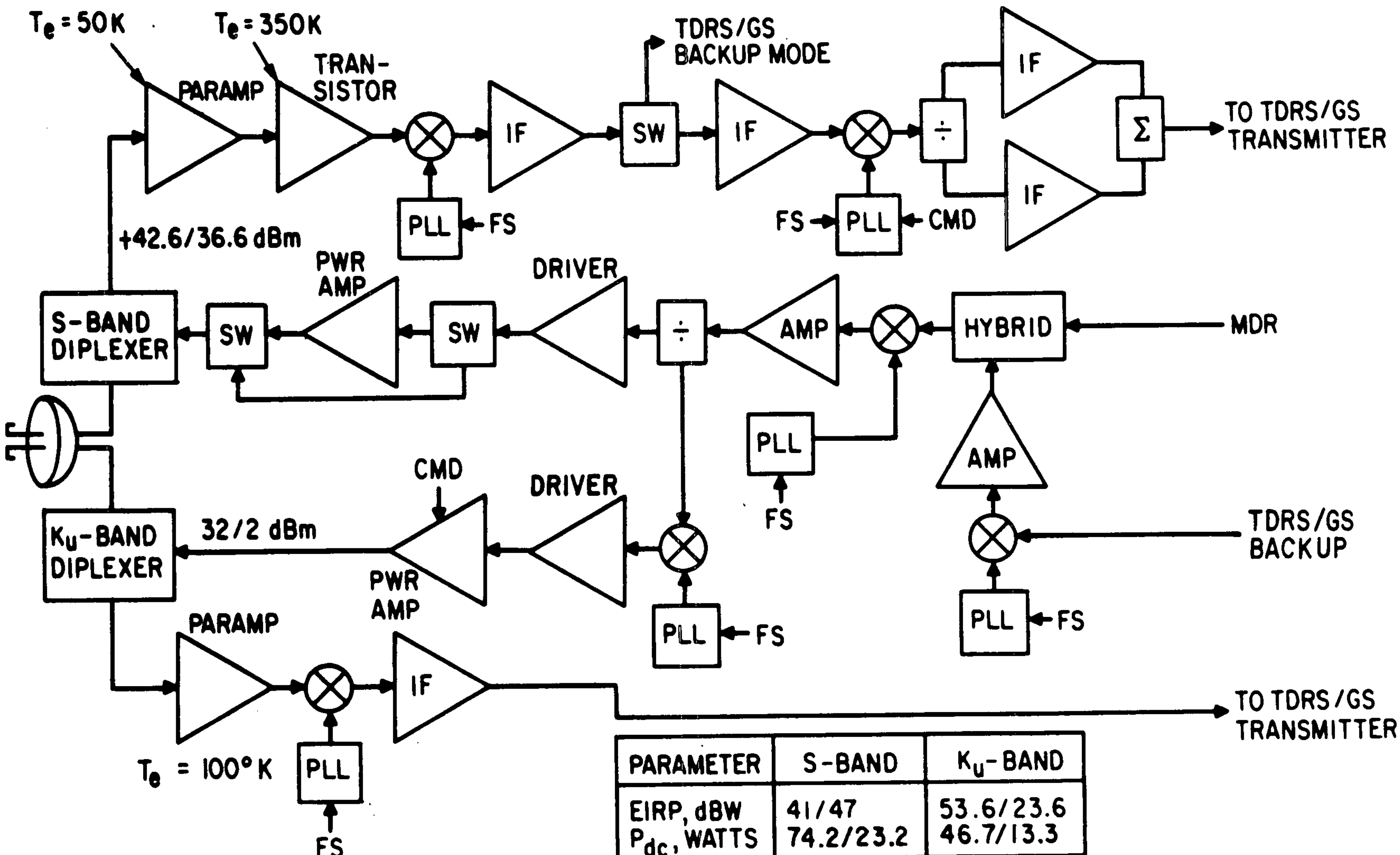


Figure 4.3-2. TDRS Telecommunication System Block Diagram



PARAMETER	S-BAND	K_u-BAND
EIRP, dBW P_{dc}, WATTS	41/47 74.2/23.2	53.6/23.6 46.7/13.3
WEIGHT, lb	63.4 INCLUDING ANTENNA	

Figure 4.3-3. MDR/HDR Transponder Design

TDRS/Ground Station Transponder

The TDRS to ground station (TDRS/GS) transponder used in the baseline system is represented in the functional diagram shown in Figure 4.3-4. The weight of signal processing modules of the subsystem is 22 pounds. Total subsystem weight including antenna is 38.6 pounds. Prime power requirements are 55.9 watts for adverse weather adequate signal penetration, and 26.1 watts for clear weather operation. Power adjustment to compensate for atmospheric weather conditions will be accomplished on the platform by means of ground command signals sent to control transmissions on a timely basis.

TDRS Tracking/Order Wire Transponder

The baseline configuration for the TDRS tracking/order wire transponder subsystem is shown in Figure 4.3-5. Transponder physical parameters are: weight, 5.3 pounds; prime power input, 7.9 watts peak and 2.0 watts average; volume of the transponder is 8.0 inches by 4.0 inches by 1.0 inch; and total weight including the antenna is 5.5 pounds.

Representative Antenna Complement

To implement the TDRS functions by means of the geosynchronous platform requires a rather complex complement of antennas on the platform. The antennas that must be accommodated to provide the service proposed in the representative TDRS used in this study are listed below:

Function	Quantity and Type
Ku-/S-band links	2 parabolic reflectors
Ku-band-only link	1 parabolic reflector
UHF forward link	4-element backfire array
VHF return link	4-element backfire array
Telemetry and tracking link	4 omni whips
S-band tracking and order wire link	1 helix whip

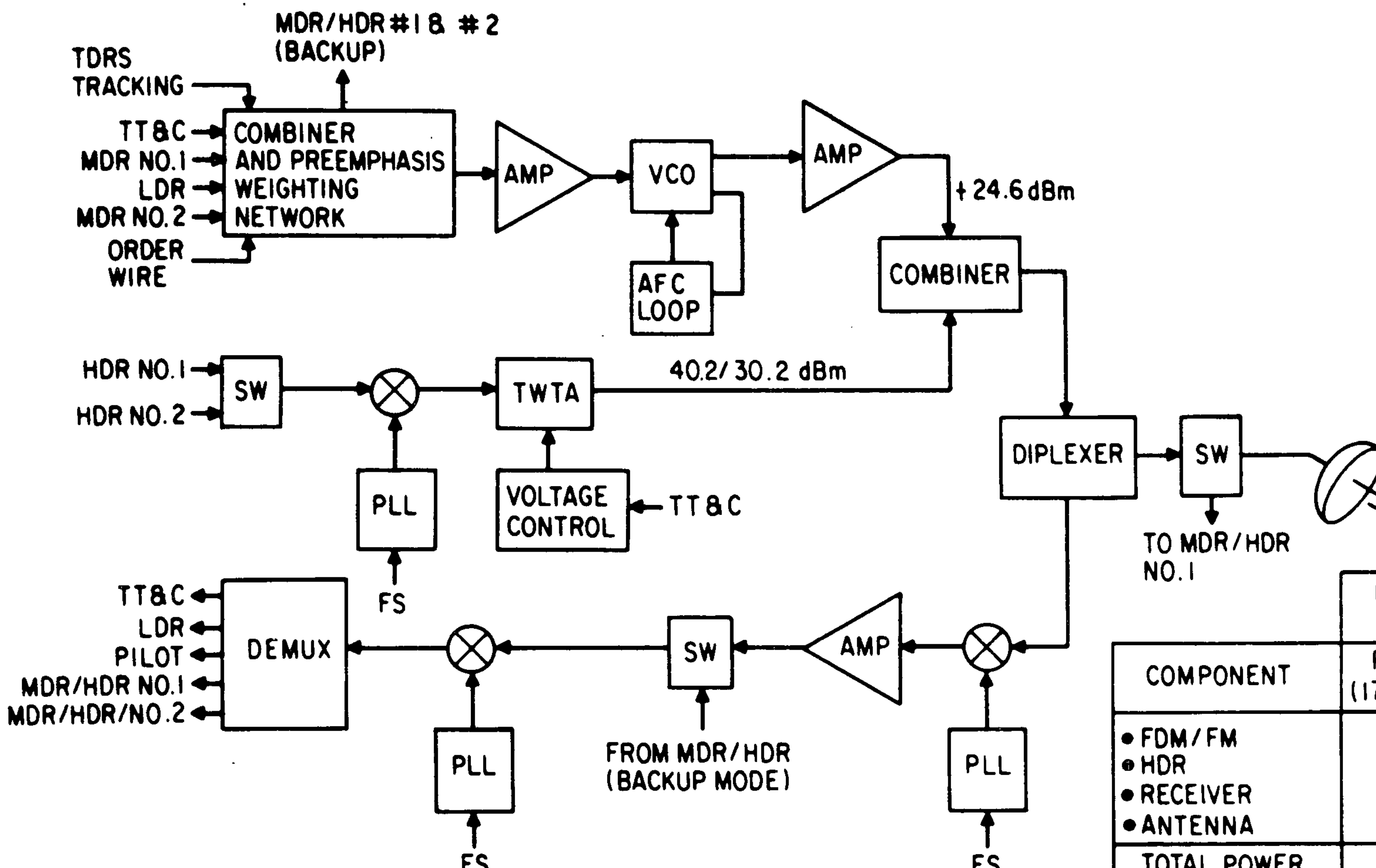
A potential layout of the TDRS antennas that are required is presented in Figure 4.3-6. The significant details pertinent to specific design features that can be implemented are shown in Figures 4.3-7, 4.3-8, 4.3-9, and 4.3-10.

PLATFORM CONFIGURATION

The TDRS design concept delineated in the preceding paragraphs is essentially one for a customized expendable satellite. Packaging concepts adaptable to the geosynchronous platform are presented in this section.

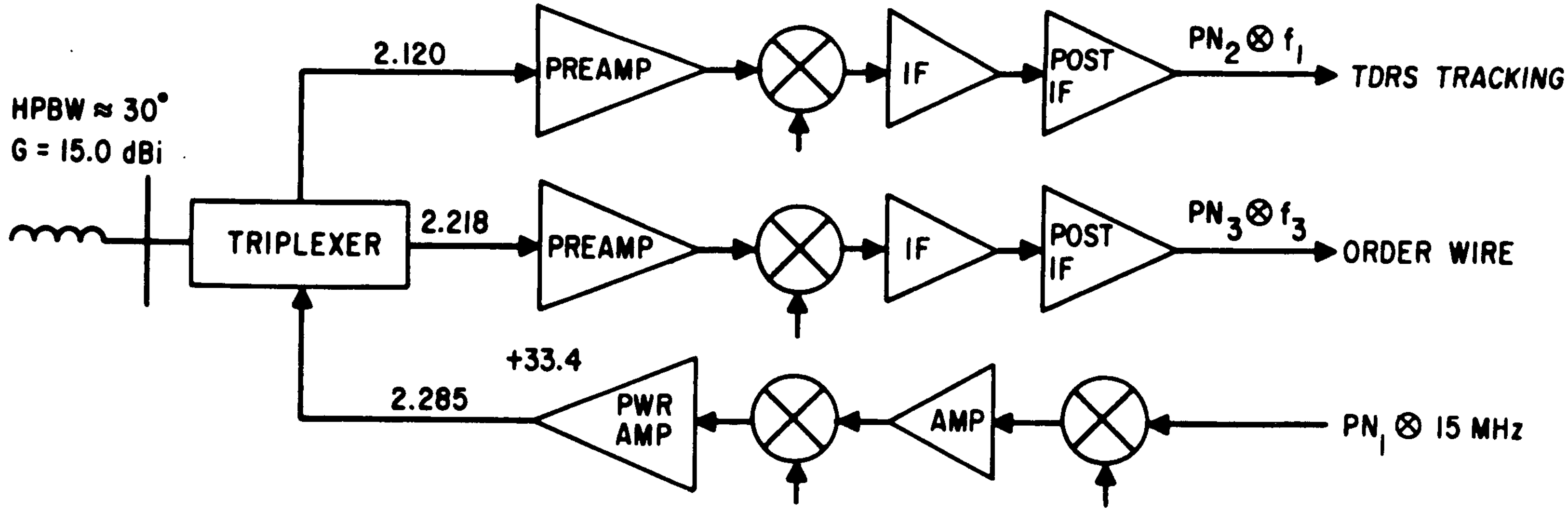
Replaceable Mission Equipment Modules

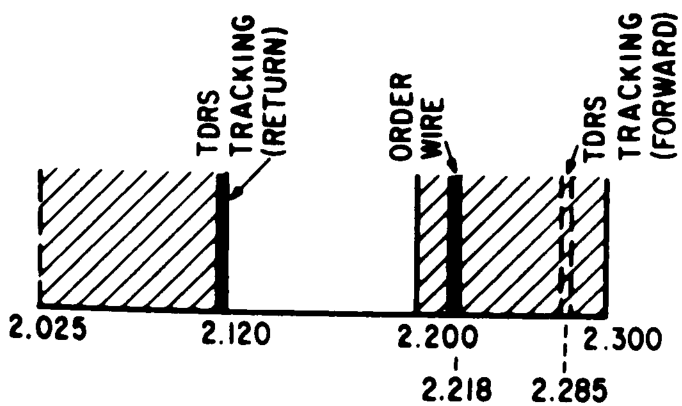
One of the primary objectives in grouping assemblies of communications equipment is to minimize the number of RF interfaces that will require on-orbit connect/disconnect operations. Therefore, it is preferred to group the entire transponder chain of assemblies or any other group of RF equipment within one



COMPONENT	PRIME POWER, WATTS RAIN (17.5dB)	CLEAR (7.5dB)
• FDM/FM	6.0	6.0
• HDR	44.6	14.8
• RECEIVER	5.3	5.3
• ANTENNA	---	---
TOTAL POWER	55.9	26.1
TOTAL WEIGHT	38.6 LB	

Figure 4.3-4. TDRS/GS Transponder Design





S-BAND CHANNEL OCCUPANCY

PARAMETER	VALUE
SIZE, inches	8" x 4" x 1"
WEIGHT, pounds	5.5
DC POWER, watts	7.9 (PEAK) 2.0 (AVG)

Figure 4.3-5. TDRS Tracking/Order Wire Transponder

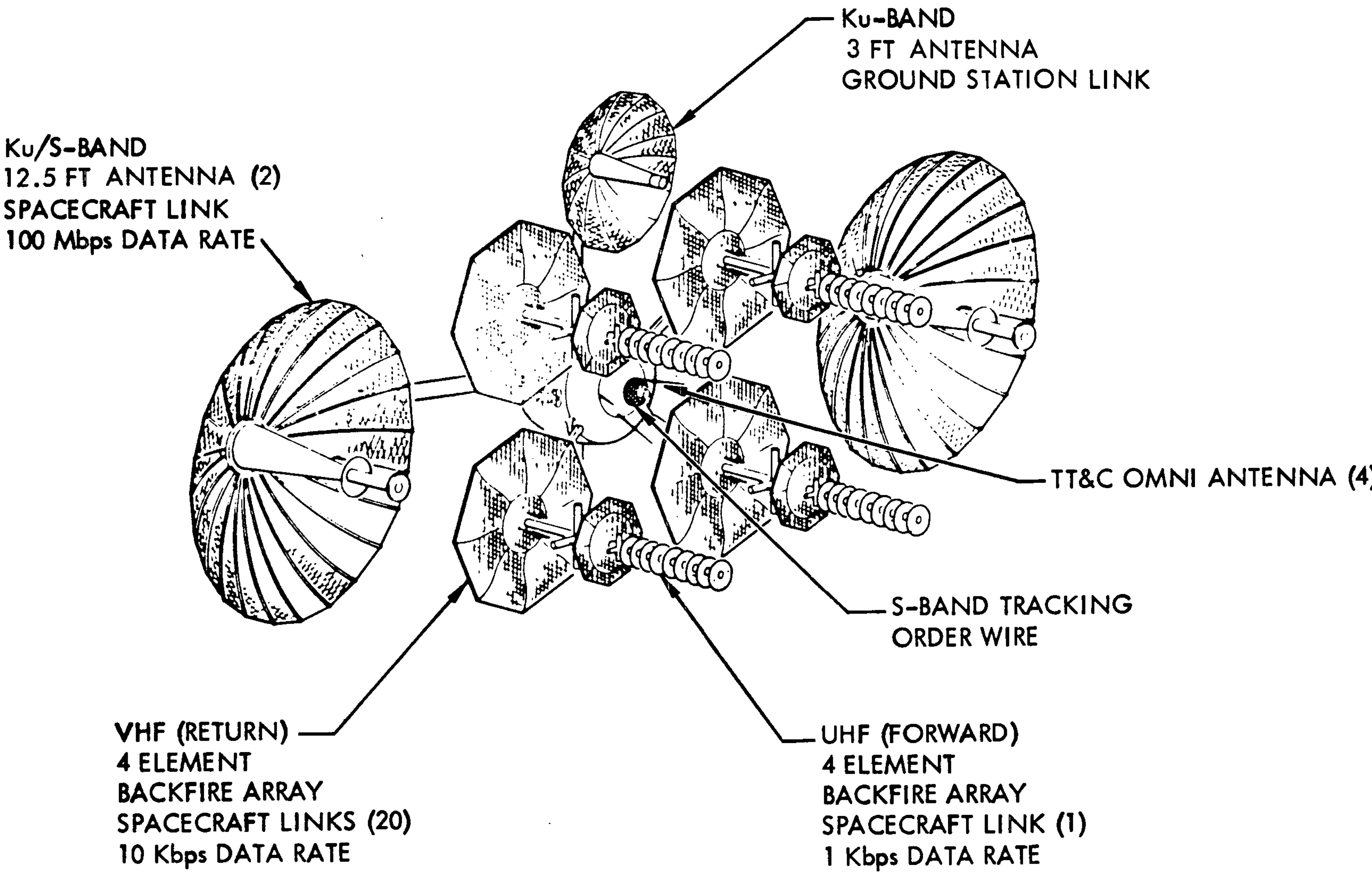
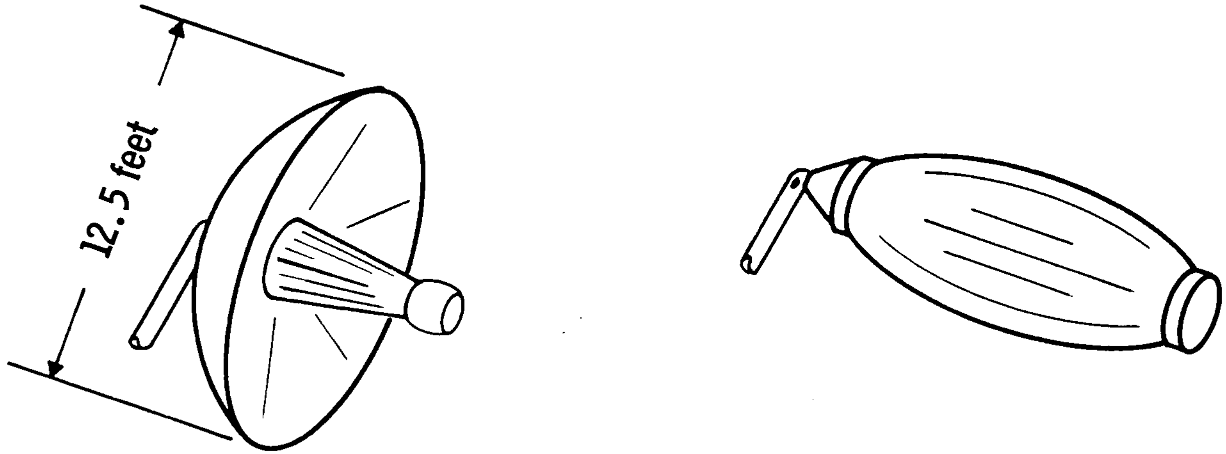


Figure 4.3-6. TDRS Antenna Configuration

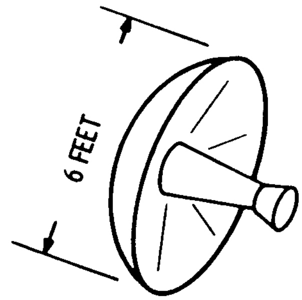


A. DEPLOYED

B. STOWED

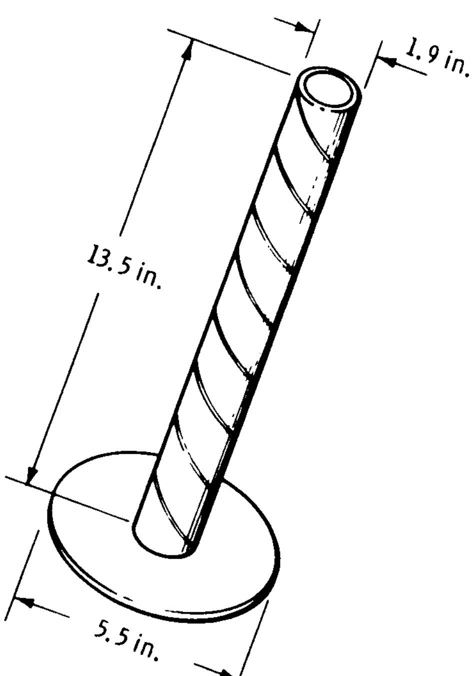
PARAMETER	S-BAND	Ku-BAND
● REFLECTOR TYPE	● DEPLOYABLE DOUBLE MESH	
● FEED DESIGN	● HELIX AT FOCAL POINT	● 4 HORN PSEUDOMONOPULSE CASSEGRAIN FEED
● HPBW	≈ 2.6°	≈ 0.4°
● GAIN – FORWARD – RETURN	35.5 dBi 36.2 dBi	53.6 dBi 52.8 dBi
● WEIGHT	39.4 lb INCLUDING SUPPORTING BOOMS	

Figure 4.3-7. MDR/HDR Dual Frequency Antenna Characteristics



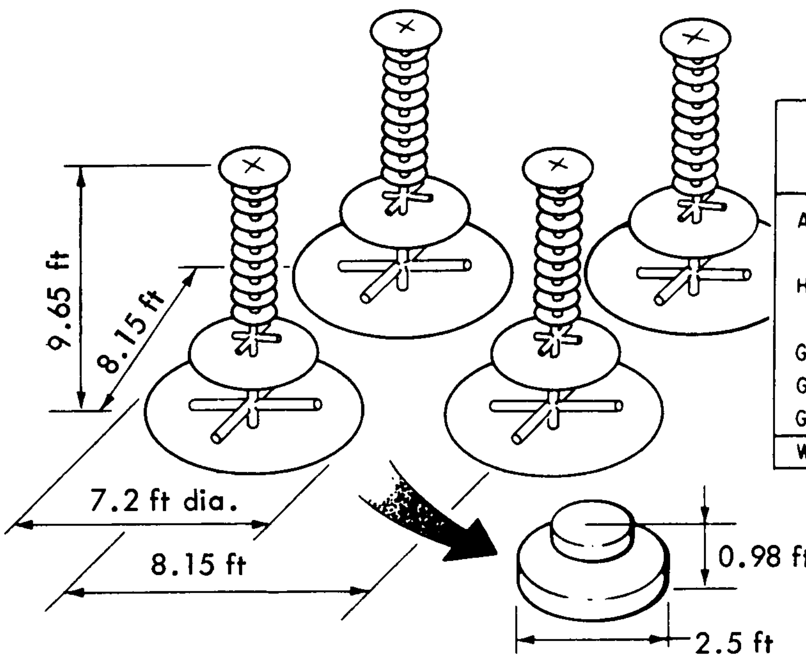
PARAMETER	CHARACTERISTICS
• REFLECTOR TYPE	• FIXED DOUBLE MESH
• FEED TYPE	• 4-HORN PSEUDOMONOPULSE CASSEGRAIN
• HPBW	• ≈ 0.8°
• GAIN - FORWARD	• 46.4 dBi
- RETURN	• 47.1 dBi
• WEIGHT	• 16.5 lb WITH BOOMS

Figure 4.3-8. TDRS/GS Antenna Characteristics



PARAMETER	CHARACTERISTICS
HPBW	30°
GAIN, dBi	
• PEAK	15.0
• 17.2° FOV	14.5
• 30° FOV	12.0
WEIGHT, LB	0.3

Figure 4.3-9. TDRS Tracking/Order Wire Antenna



A. *DEPLOYED* B. *STOWED*

PARAMETERS	VHF		UHF	
	AGIPA	F-FOV	F-FOV	STEERED
ANTENNA TYPE	QUAD ARRAY OF SHORT BACK-FIRE ELEMENTS		QUAD ARRAY OF DISC-ON-RODS	
HPBW-ELEMENT	46°	46°	31°	31°
-ARRAY	23°	23°		9°
GAIN-PEAK	17.4 dBi	17 4 dBi	15.0 dBi	21.0 dBi
GAIN-26° FOV	16.5 dBi	13.3 dBi	13.0 dBi	19.0 dBi
GAIN-30° FOV	16.2 dBi	11.6 dBi	12 0 dBi	18.0 dBi
WEIGHT	31.6 lb INCLUDING BOOMS			

Figure 4.3-10. Dual Frequency (VHF-UHF) LDR Quad Array

module. In the case of the TDRS, a logical grouping of equipment is the MDR/HDR assemblies, the LDR assemblies, and the TDRS/GS assemblies combined with the tracking beacon order wire and frequency source assemblies. Figures 4.3-11, 4.3-12 and 4.3-13 depict packaging concepts for the three modules. All assemblies readily fit within the standardized 24 inches by 20 inches by 24 inches replaceable module envelope.

TDRS Platform Layout

Figure 4.3-14 illustrates a completely assembled TDRS platform. Also shown is the approximate configuration in the space shuttle cargo bay. The support systems are contained in the common support module. Only 3 of the 12 available equipment compartments in the mission equipment ring are used. Commonality of equipment across all platforms was the principal reason for this apparent overdesign. It is believed that this commonality is more cost efficient than developing a customized mission equipment configuration for the TDRS. Table 4.3-4 summarizes the weights and power requirements for the TDRS.

Table 4.3-4. TDRS Platform Summary

Item	Weight (lb)		Power (watts)	
	Subtotal	Total	Subtotal	Total
Structure		439		--
Primary	100			
Secondary	300			
Thermal protection	39			
Mission Equipment		223		273
UHF/VHF transponder	15		115	
AGIPA antenna	32			
MDR/HDR transponder (2)	48		100	
Antennas (2)	79			
TDRS/GS transponder	26		56	
Antenna	17			
Auxiliary equipment	6		2	
Structure		500		--
Subsystems		1489		218
Totals		2651		491

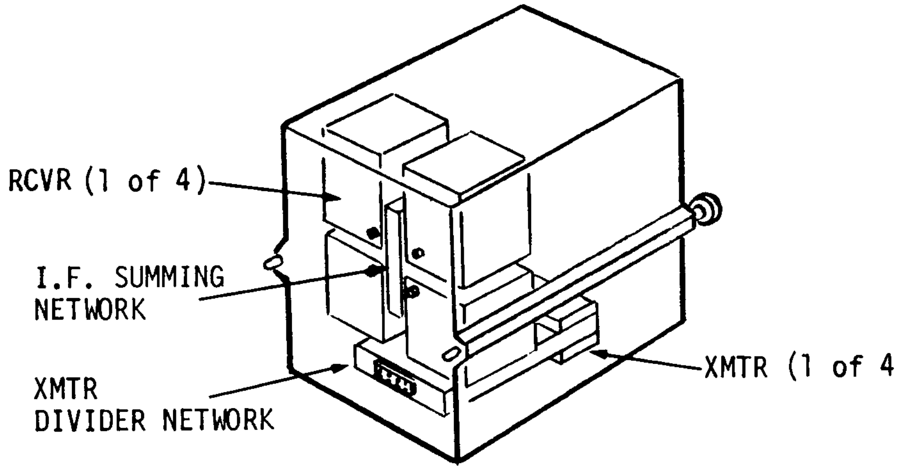


Figure 4.3-11. LDR Transponder Assembly

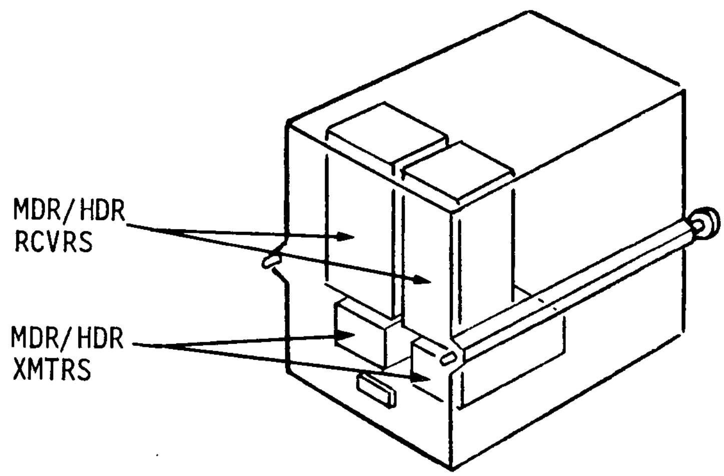


Figure 4.3-12. MDR/HDR Transponder Assembly

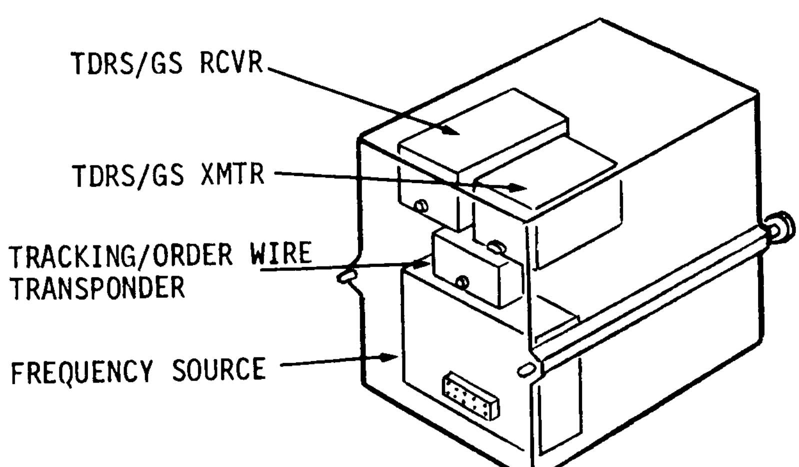


Figure 4.3-13. TDRS Auxiliary Equipment

IMPLEMENTATION PLAN

The implementation plan is straightforward. During the space shuttle era, TDRS satellites launched in the late 1970's are scheduled to be replaced in 1983 with new expendable satellites. With a geosynchronous platform program, the TDRS platforms would be delivered in 1983. With the on-orbit servicing provisions incorporated in the platform design, replacement of TDRS space elements as currently scheduled for 1989 would not be required.

4.4 DATA RELAY PLATFORM SYNTHESIS FOR THE NEW TRAFFIC MODEL

Design of the data relay platforms for the new traffic model follows the same general pattern as the baseline mode. The new traffic model design must accommodate significantly larger traffic in all regions. Large traffic growths are projected for the developing countries. This growth is supported by increasing the basic platform channel capacity and by providing more platforms in each region. Section 5.3 of Volume IV, Part 2, defined the numbers of platforms per region and the regional distribution pattern. Section 4.1 of this volume defined the design criteria that allowed support of this distribution.

Domsat platforms with a total capacity of 216 transponder channels and Intelsat platforms of 192 channels are the basic units used in each region. The major variation in individual platforms evolves from the necessity to use different antenna pattern arrangements in each region. All countries must be supported by sufficient Domsat channels for both "all-country broadcast" service and intra-country traffic of all types. This determines the necessity for a varying number of spot beams (two-degree) and area coverage beams (five-degree and shaped). Platform configurations vary from four antennas and five beams in Region I to 16 antennas and 22 beams in Region IV.

Intelsat platforms using only C-band and K_{HI}-band are designed to ensure spot coverage of international traffic rates. In some cases, broader coverage of continental areas with five-degree C-band beams is necessary to support all of the countries on a continent. Intelsat platform configurations vary from 8 antennas and 14 beams in Region I to 11 antennas and 22 beams for Regions II and III.

Packaging of C- and K_{LO}-band transponder equipment is essentially the same as that from the baseline traffic model. Both traffic model platforms utilize 24-channel transponder systems on these bands. In order to accommodate the large number of K_{HI}-band channels (168) and the major beam switching matrix, a new integrated package design was necessary.

A unique integrated package was postulated to service an 84-channel by 15-beam (maximum) distribution system. By utilizing a unique configuration of PIN diode switching matrices, multiplexers and combiners, this large total transponder and switching system was compressed into a package 20 inches by 24 inches by 36 inches. All RF inputs and outputs were placed on the rear 20-inch by 24-inch face of the package to enable removal for servicing. This configuration is 12 inches longer than a standard module, but fits easily into the overall platform structure. Two such packages are required for a 168-channel K_{HI}-band system; one for 84 odd channels and one for the 84 even channels. Odd and even channels are of opposite orthogonal polarization.

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Because of the multiple platform arrangement in each region, it is possible to provide gradual growth in regional traffic capability.

REGIONAL EQUIPMENT COMPLEMENT

Domsat Platforms

In Section 5.3, Volume IV, Part 2, the distribution of traffic by country and region was determined. Table 4.4-1 shows this distribution. The next step was to determine the distribution of this traffic among the platforms and frequency bands allocated in accordance with the maximum available as defined in the Design Criteria section: C-band = 24 channels, K_{LO}-band = 24 channels, and K_{HI}-band = 168 channels. No frequency reuse was applied on any platform--except for the overlapping orthogonally polarized channels already accounted for in the defined maximum number of channels.

A set of charts was used to define the distribution of channels to each platform in each region. Several ground rules were used to provide proper coverage of continents and countries. These were:

1. Ensure that any country has the capability to provide a service that allows broadcast from one ground station to all others in that country with use of a single channel only.

2. Ensure that all channels on a given platform have the capability to service any part of the country in its service area.

3. Attempt, wherever possible, to divide the traffic among platforms in the same region so that complete solar outages can be avoided.

4. Provide the flexibility to allow traffic rerouting in peak demand periods for specific traffic rates.

Each of these items has a different impact on the platform antenna and beam configuration. To accomplish the broadcast service of Item 1, it is necessary to use wide beamwidth patterns (five degrees) associated with C-band. This is especially true in large area countries. In some cases, C-band shaped beams are necessary for this service.

Country-wide communications place a constraint on the minimum number of narrow beams for a single country that are utilized on a single platform. If four beams are necessary to cover the country, they must all be on the same frequency band and available on the same platform. Again, this constraint is a problem in the larger countries only.

Distribution of a single country traffic to avoid solar outages is only a problem where the country's traffic is low and does not warrant the complexity associated with the penalty of extra antennas and beams on a given platform.

Use of the quasi-TDMA antenna beam switching system provides the flexibility needed not only for traffic rerouting but enables compliance with the ground rule for country-wide communications.

Table 4.4-1. Regional Traffic Distribution Requirements

PLATFORM LOCATION (DEGREES)	SERVICE AREA	ORBITAL LIMITS (DEGREES)	(A) NUMBER OF C-BAND SATELLITES	(B) NUMBER OF CHANNELS (A) x 24	REQUIRED NUMBER OF PLATFORMS (B) ÷ 216	TOTAL REGION PLATFORM CAPABILITY
REGION I 172E			18	432	2	432
	JAPAN AUSTRALIA USSR (EAST) INDOCHINA EAST INDIES	75 TO 170E 90 TO 120E	7 1 4 2 4			
REGION II 68E			52	1248	6	1296
	INDIA CHINA	16 TO 138E 60 TO 148E	17 35			
REGION III 6W			42	1008	5	1080
	EUROPE USSR (WEST) AFRICA	30W TO 55E 0 TO 85E 15W TO 57E	23 8 11			
REGION IV 110W			34	816	4	864
	USA (CANADA) SA	138W TO 53W 110 TO 10W	21 13			

Tables 4.4-2 through 4.4-5 illustrate the traffic distribution for the Domsat service. An iterative process involving geographical beam placement was necessary to result in these tables.

A similar procedure as that used for the baseline traffic model was performed with beam layouts on regional maps. The 5-degree and 2-degree beam overlays developed in Volume III were used with further support from the computer program. The results for the Domsat platforms in each region are shown in Figures 4.4-1 through 4.4-7. In some cases, C-band 5-degree and K_{LO}/K_{HI}-band 2-degree beams are shown on separate charts. This is to relieve the complexity of a given chart. Each chart identifies 5-degree mask angle limits of interest to ensure that coverage of specific areas can be provided from the various platform locations.

The beam patterns are shown from one location. In many cases, overlapping beam coverage is supplied from other platforms. Although the pattern would actually be slightly different, it is not shown in the interest of simplicity. Reference to the tables and geographical patterns provides full understanding of beam distribution on each satellite.

Intelsat Platforms

For the Intelsat traffic distribution, the main concern was to ensure that all countries within the Intelsat regions could communicate with each other. Arbitrary locations of spot beams were chosen to satisfy the major international traffic routes. K_{HI}-band beams were used for this application. Broader 5-degree C-band beams were used to cover other continental countries to provide capability for a limited amount of traffic to these areas. The traffic model showed the following traffic distribution:

Region	Area	C-Band 24-Channel Reference	Number of Channels	Equivalent Platforms	Capacity
I	Pacific Ocean	15	360	2	384
II	Indian Ocean	18	432	2	384
III	Atlantic Ocean	11	264	2	300

As previously explained, the need for 48 more channels in Region II can be solved by utilizing the same frequencies in widely separated beams at K_{HI}-band or the excess Domsat capacity. All other design parameters, except for the fact that K_{LO}-band is not used, are the same as the Domsat items. Figures 4.4-8 through 4.4-12 illustrate the regional beam pattern layouts to support the new traffic model for Intelsat.

Table 4.4-2. Region I Domsat Platform Traffic Distribution

POSITION	167 DEGREES			177 DEGREES			TOTAL APPLIED	NUMBER REQUIRED
PLATFORM	I			II				
BAND	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}		
TOTAL	24	24	168	24	24	168		
NUMBER OF BEAMS	1	2	3	2	2	4		
NUMBER OF ANTENNAS	1	1	2	1	1	2		
AREA CHANNELS								
JAPAN	-	-	72	-	-	96	168	168
USSR (EAST)	-	-	96	-	-	-	96	96
INDOCHINA	-	-	-	-	-	48	48	48
EAST INDIES*	24	24	-	-	24	24	96	96
OCEANA**	-	-	-	24	-	-	24	24
BEAMS								
JAPAN	-	-	1	-	-	1		
USSR (EAST)	-	-	2	-	-	-		
INDOCHINA	-	-	-	-	-	3		
EAST INDIES	1	2	-	-	2	-		
OCEANA	-	-	-	2	-	-		
TOTAL CHANNELS							432	432

*INCLUDES PHILLIPINES

**AUSTRALIA/NEW ZEALAND

Table 4.4-3. Region II Domsat Platform Traffic Distribution

POSITION (DEG)	55E			65E			75E			80E			85E			90E			TOTAL APPLIED	NUMBER REQUIRED
PLATFORM	I			II			III			IV			V			VI				
BAND	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}		
TOTAL	24	24	168	24	24	168	24	24	168	24	24	168	24	24	168	24	24	168		
NO. OF BEAMS	1	3	3	1	1	6	1	3	3	1	1	6	1	1	6	1	1	6		
NO. OF ANTENNAS	1	2	2	1	1	3	1	2	2	1	1	3	1	1	3	1	1	3		
AREA																				
CHANNELS																				
CHINA	-	-	-	24	24	168	-	-	-	24	24	168	24	24	168	24	24	168	864	840
INDIA	24	24	168	-	-	-	24	24	168	-	-	-	-	-	-	-	-	-	432	408
BEAMS																				
CHINA	-	-	-	1	1	6	-	-	-	1	1	6	1	1	6	1	1	6		
INDIA	1	3	3	-	-	-	1	3	3	-	-	-	-	-	-	-	-	-		
TOTAL CHANNELS																			1296	1248

Table 4.4-4. Region III Domsat Platform Traffic Distribution

POSITION (DEG)	10W			0			5E			10E			15E			TOTAL APPLIED	NUMBER REQUIRED
PLATFORM	I			II			III			IV			V				
BAND	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}		
TOTAL	24	24	168	24	24	168	24	24	168	24	24	168	24	24	168		
NO. OF BEAMS	1	4	4	3	4	4	3	4	13	3	3	16	1	3	10		
NO. OF ANTENNAS	1	2	2	2	2	2	2	2	7	2	2	8	1	2	5		
AREA																	
CHANNELS																	
EUROPE	24	24	168	-	24	168	-	24	48	-	-	50	-	-	66	596	552
USSR	-	-	-	-	-	-	-	-	72	-	24	-	24	-	72	192	192
AFRICA	-	-	-	24	-	-	24	-	48	24	-	118	-	24	30	292	264
BEAMS																	
EUROPE	1	4	4	-	4	4	-	4	4	-	-	4	-	-	4		
USSR	-	-	-	-	-	-	-	-	3	-	3	-	1	-	3		
AFRICA	-	-	-	3	-	-	3	-	6	3	-	12	-	3	3		
TOTAL CHANNELS																1080	1008

Table 4.4-5. Region IV Domsat Platform Traffic Distribution

POSITION (DEGREES)	120W			115W			110W			105W			TOTAL APPLIED	NUMBER REQUIRED
PLATFORM	I			II			III			IV				
BAND	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}	C	K_{LO}	K_{HI}		
TOTAL	24	24	168	24	24	168	24	24	168	24	24	168		
NUMBER OF BEAMS	7	6	15	7	6	15	7	6	15	7	4	15		
NUMBER OF ANTENNAS	5	3	8	5	3	8	5	3	8	5	2	8		
AREA														
CHANNELS														
USA	9	10	84	9	10	84	9	10	84	9	-	90	408	408
CANADA	4	14	-	4	14	-	4	14	-	4	14	-	72	72
MEXICO	6	-	-	6	-	-	6	-	-	6	-	-	24	24
SOUTH AMERICA	5	-	84	5	-	84	5	-	84	5	-	78	350	312
BEAMS														
USA	3	2	5	3	2	5	3	2	5	3	-	5		
CANADA	1	4	-	1	4	-	1	4	-	1	4	-		
MEXICO	1	-	-	1	-	-	1	-	-	1	-	-		
SOUTH AMERICA	2	-	10	2	-	10	2	-	10	2	-	10		
TOTAL CHANNELS													854	816

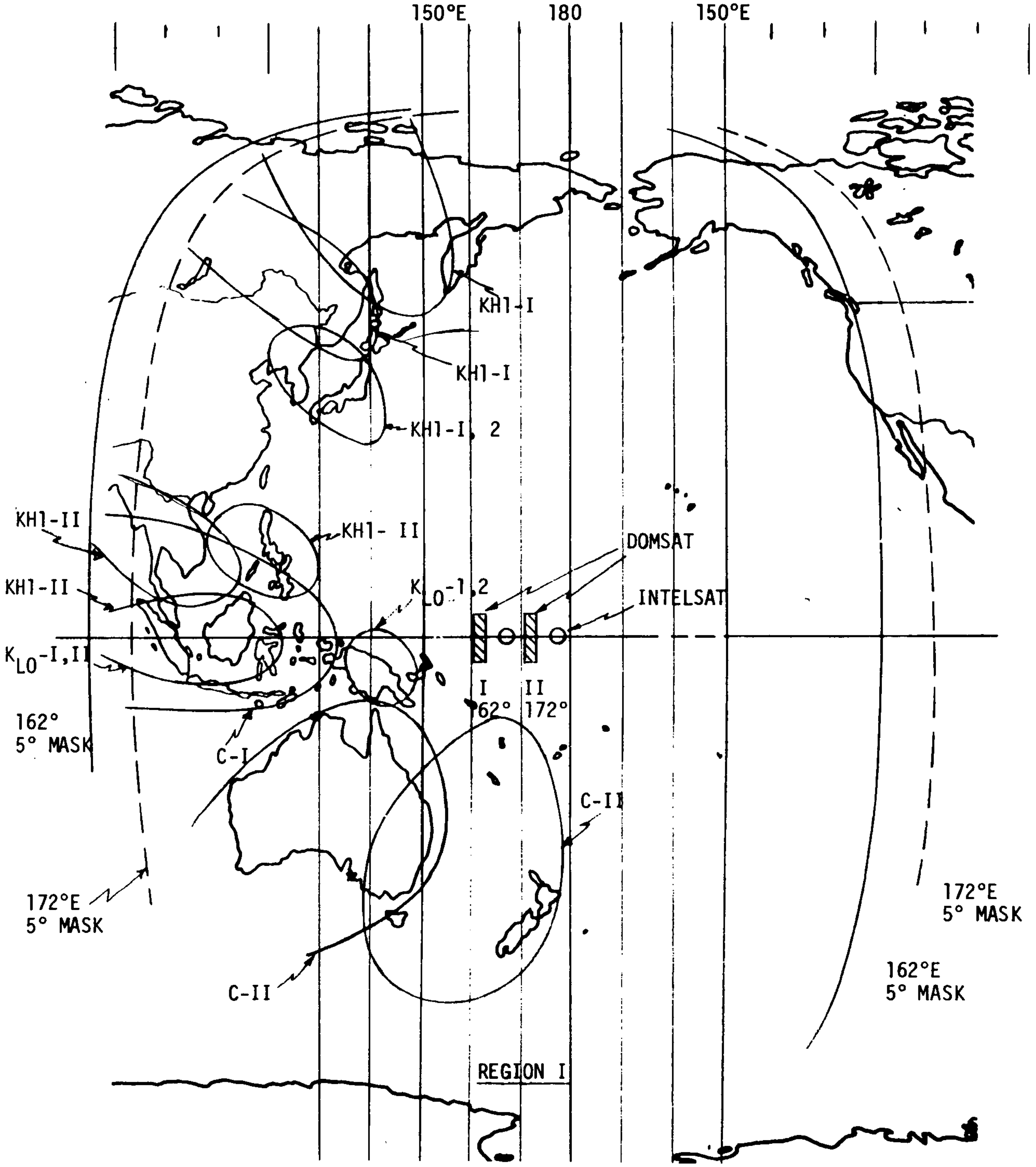


Figure 4.4-1. Domsat Region I Platform Antenna Beam Pattern Layout

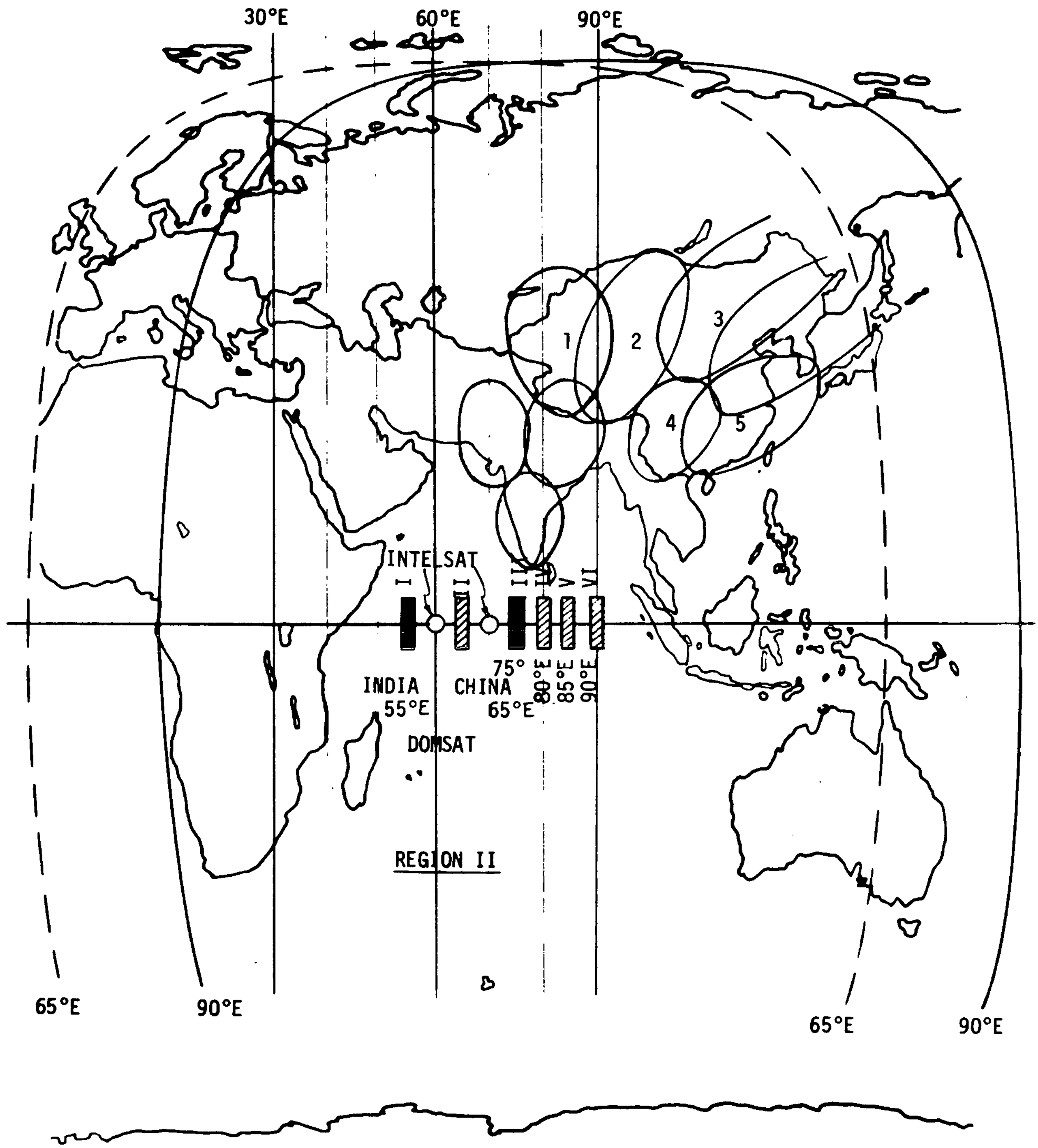


Figure 4.4-2. Domsat Region II Platform 2° Spot Beam Pattern Layout

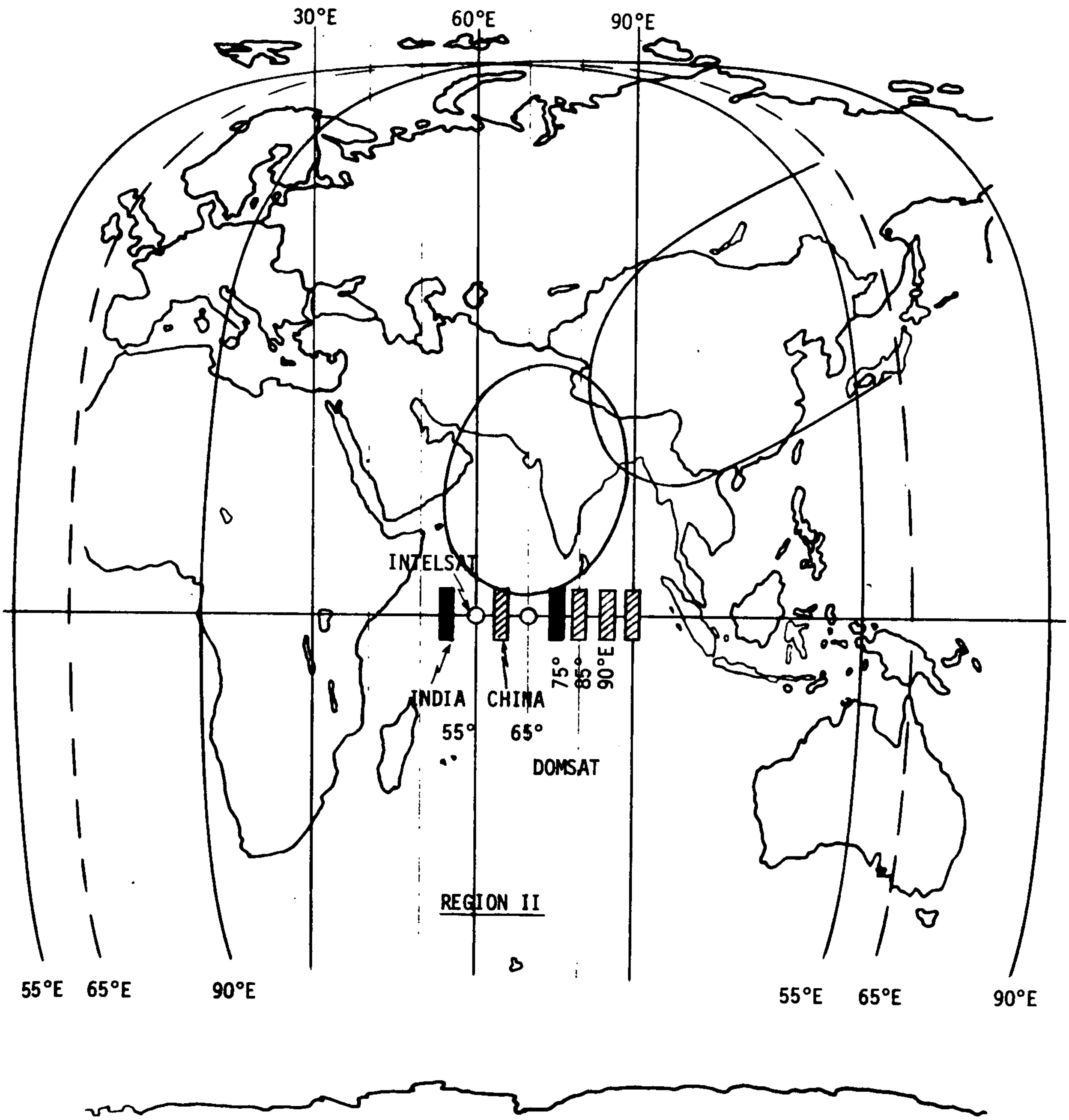


Figure 4.4-3. Domsat Region II Platform C-Band 5° Beam Pattern Layout

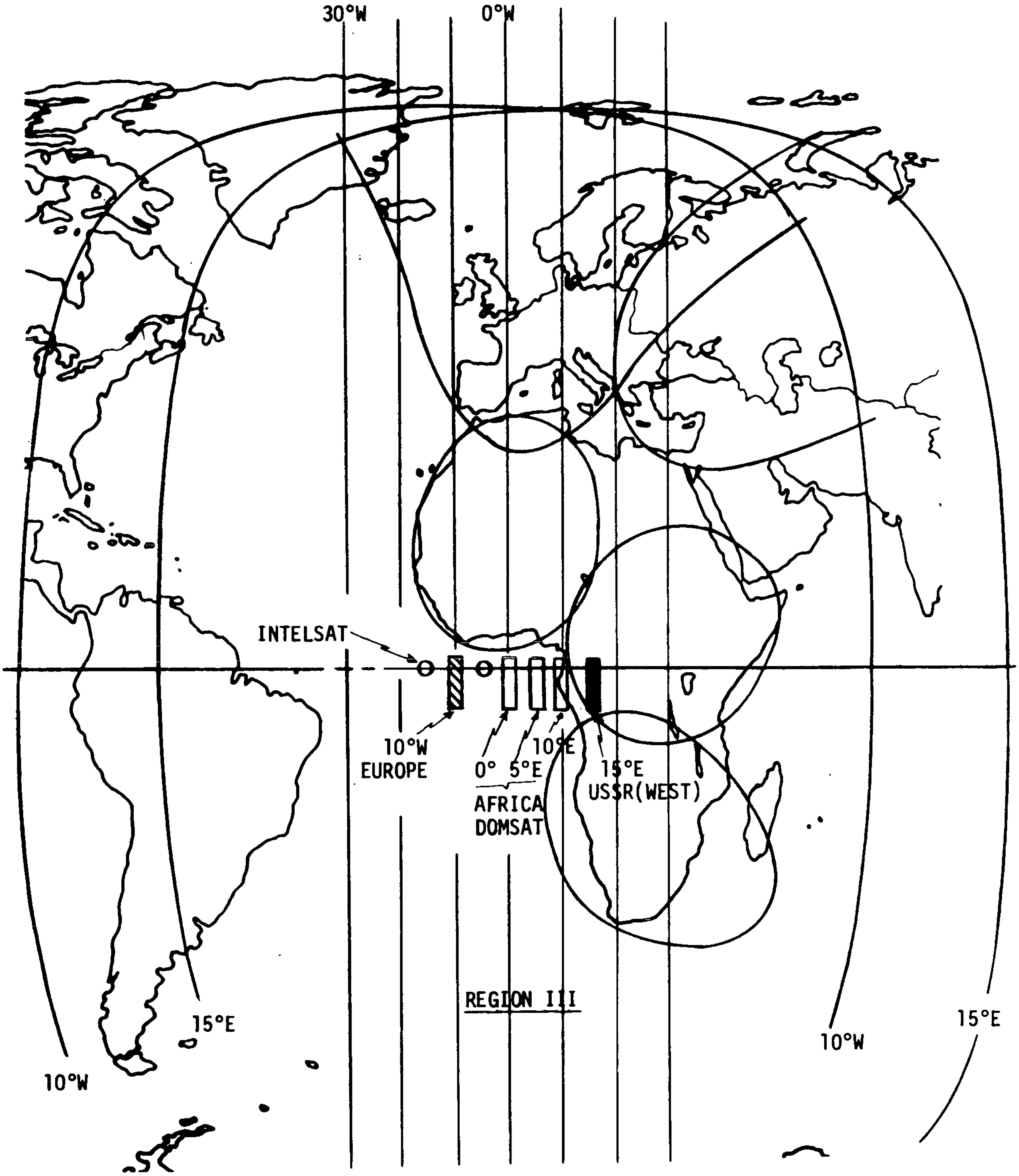


Figure 4.4-4. Domsat Region III Platform C-Band 5° Beam Pattern Layout

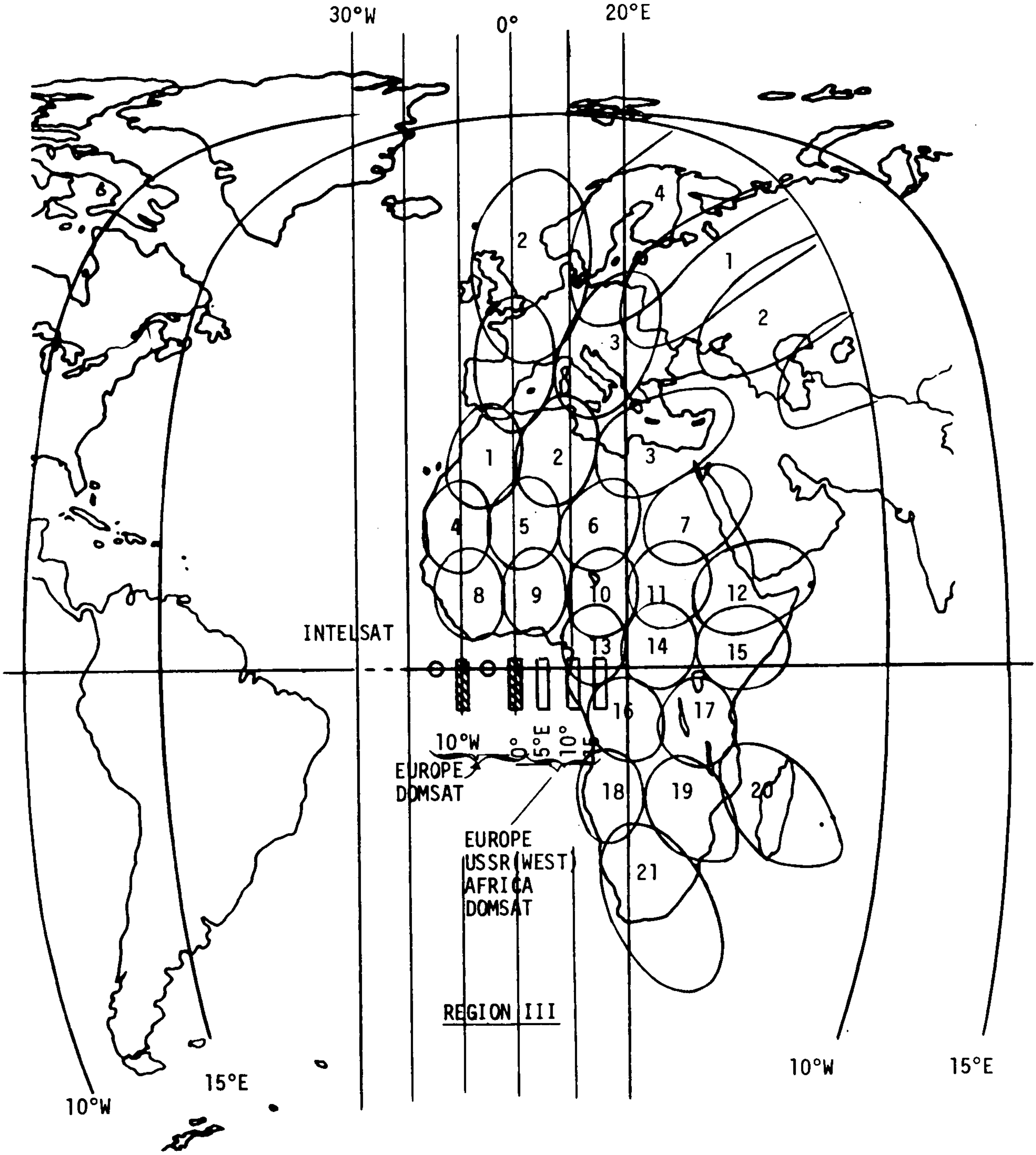


Figure 4.4-5. Domsat Region III Platform 2° Spot Beam Pattern Layout

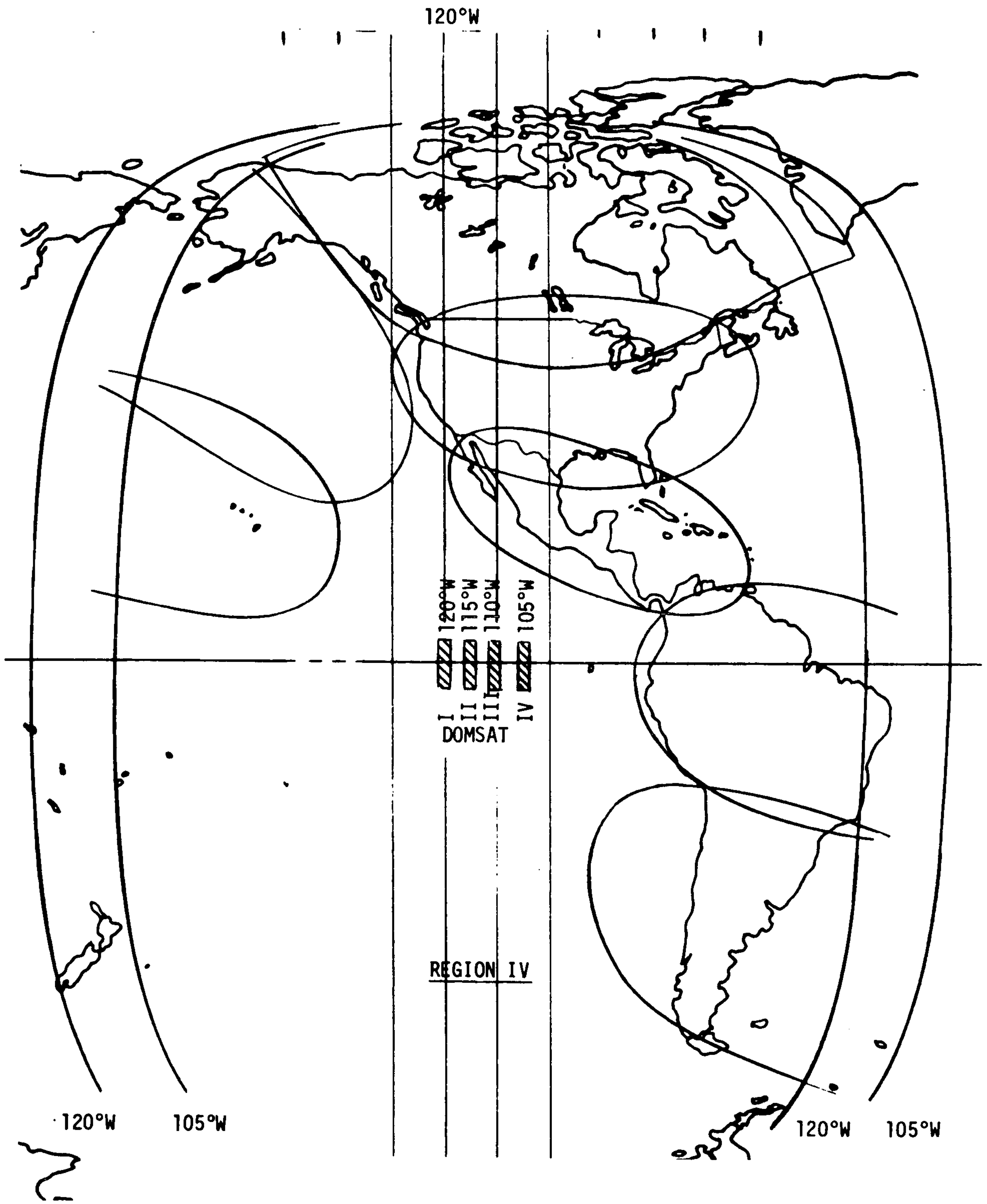


Figure 4.4-6. Domsat Region IV C-Band 5° Beam Pattern Layout

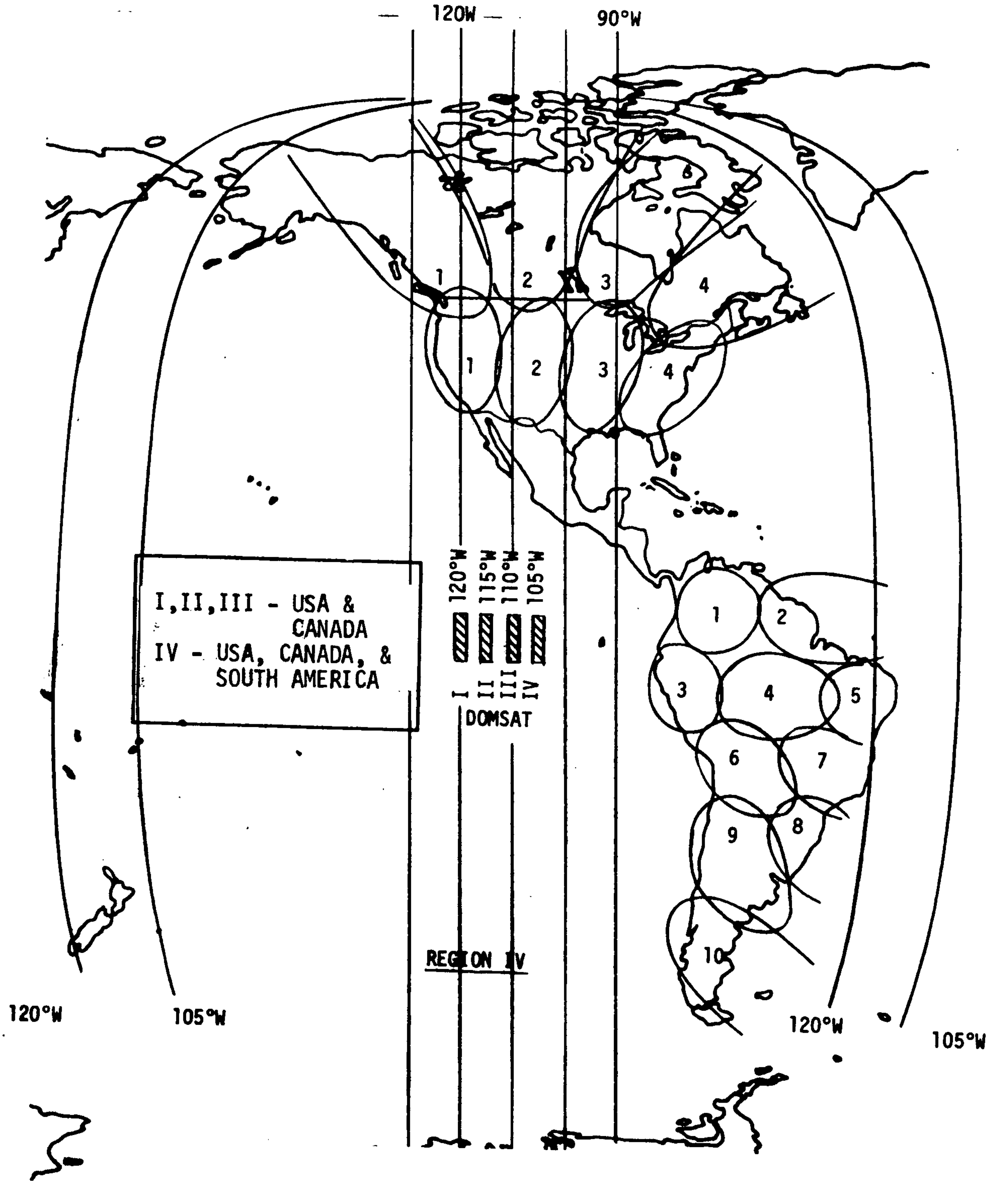


Figure 4.4-7. Domsat Region IV 2° Spot Beam Pattern Layout

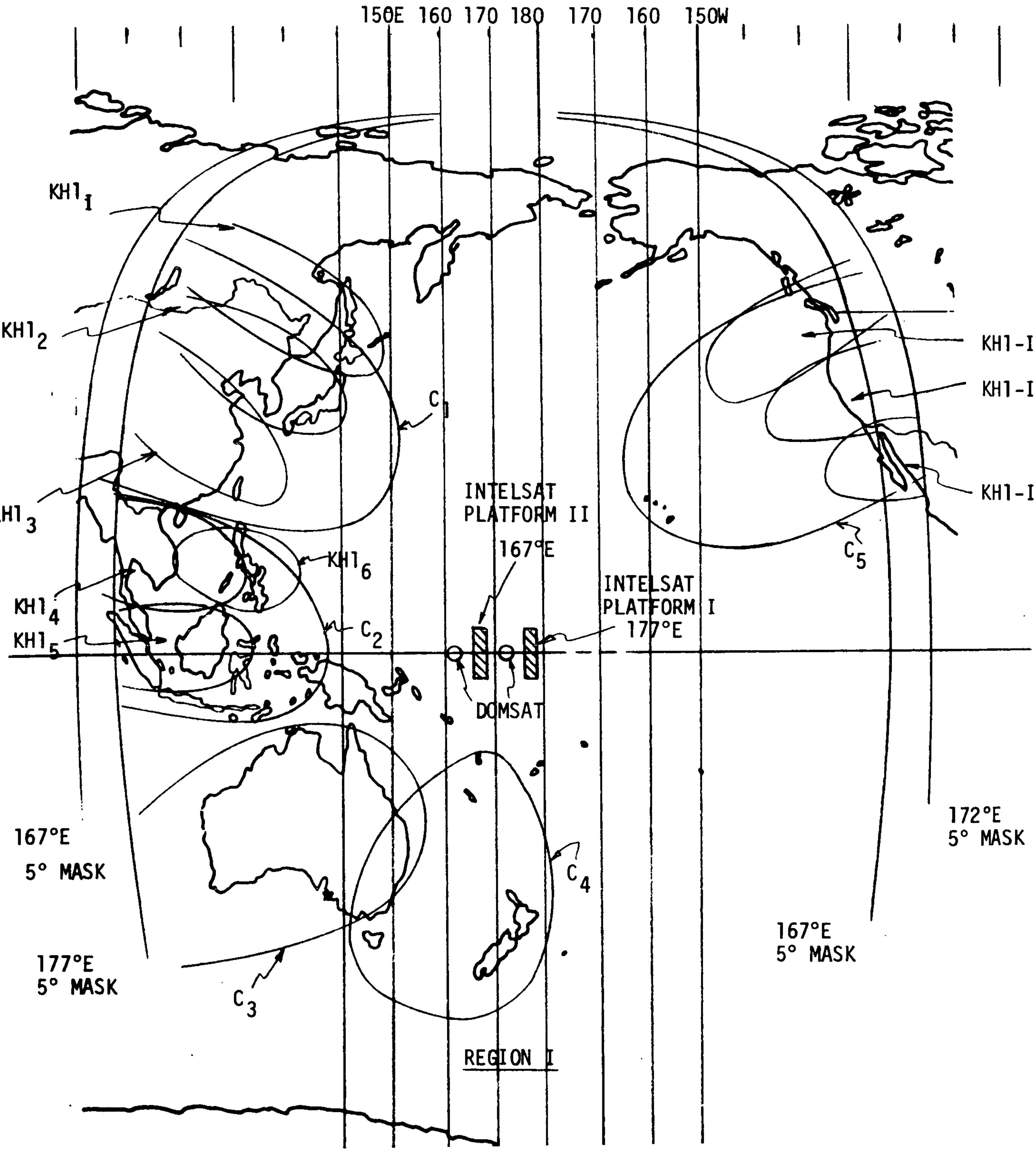


Figure 4.4-8. Intelsat Region I Platform Antenna Beam Pattern Layout

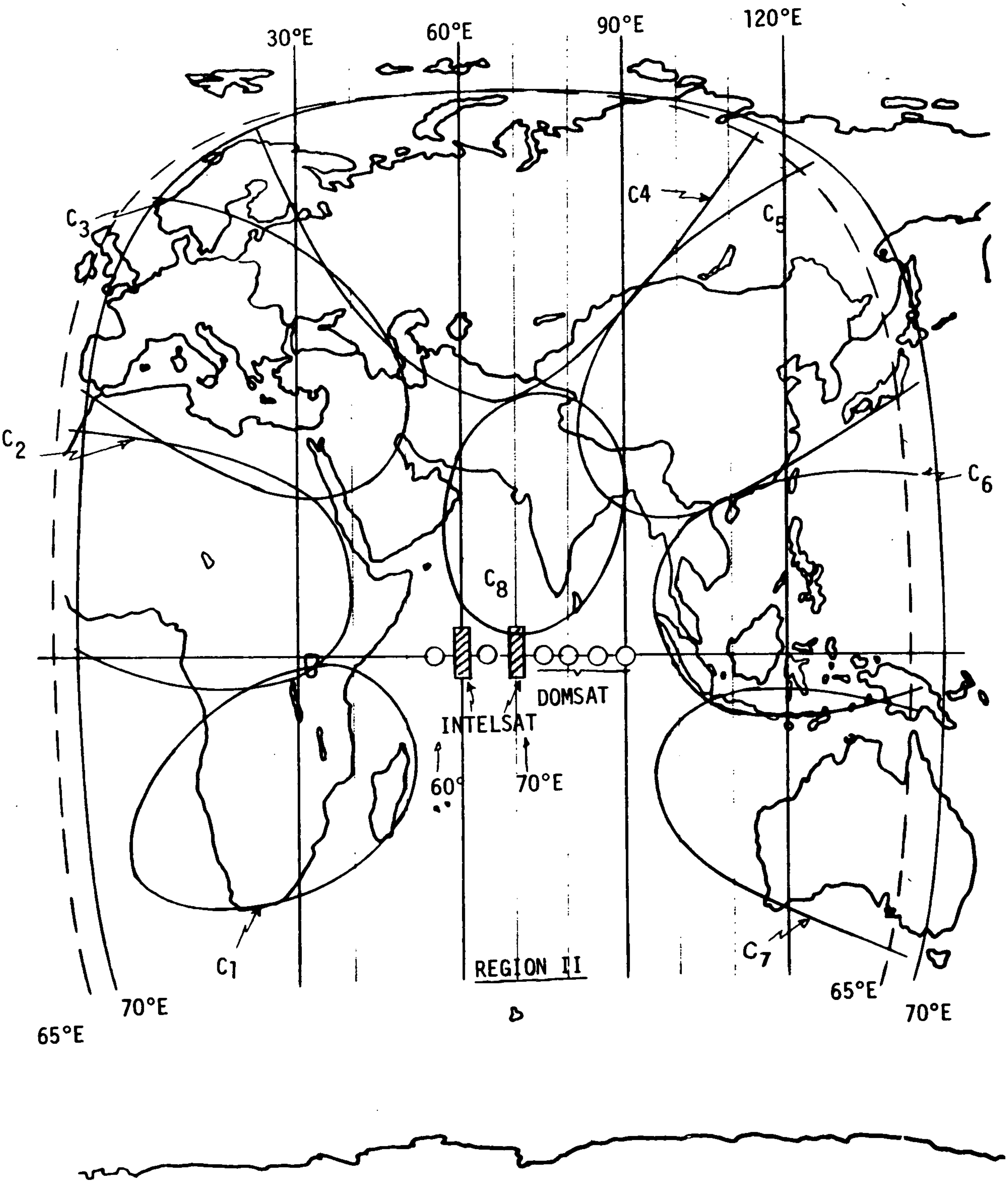


Figure 4.4-9. Intelsat Region II Platform C-Band 5° Beam Pattern Layout

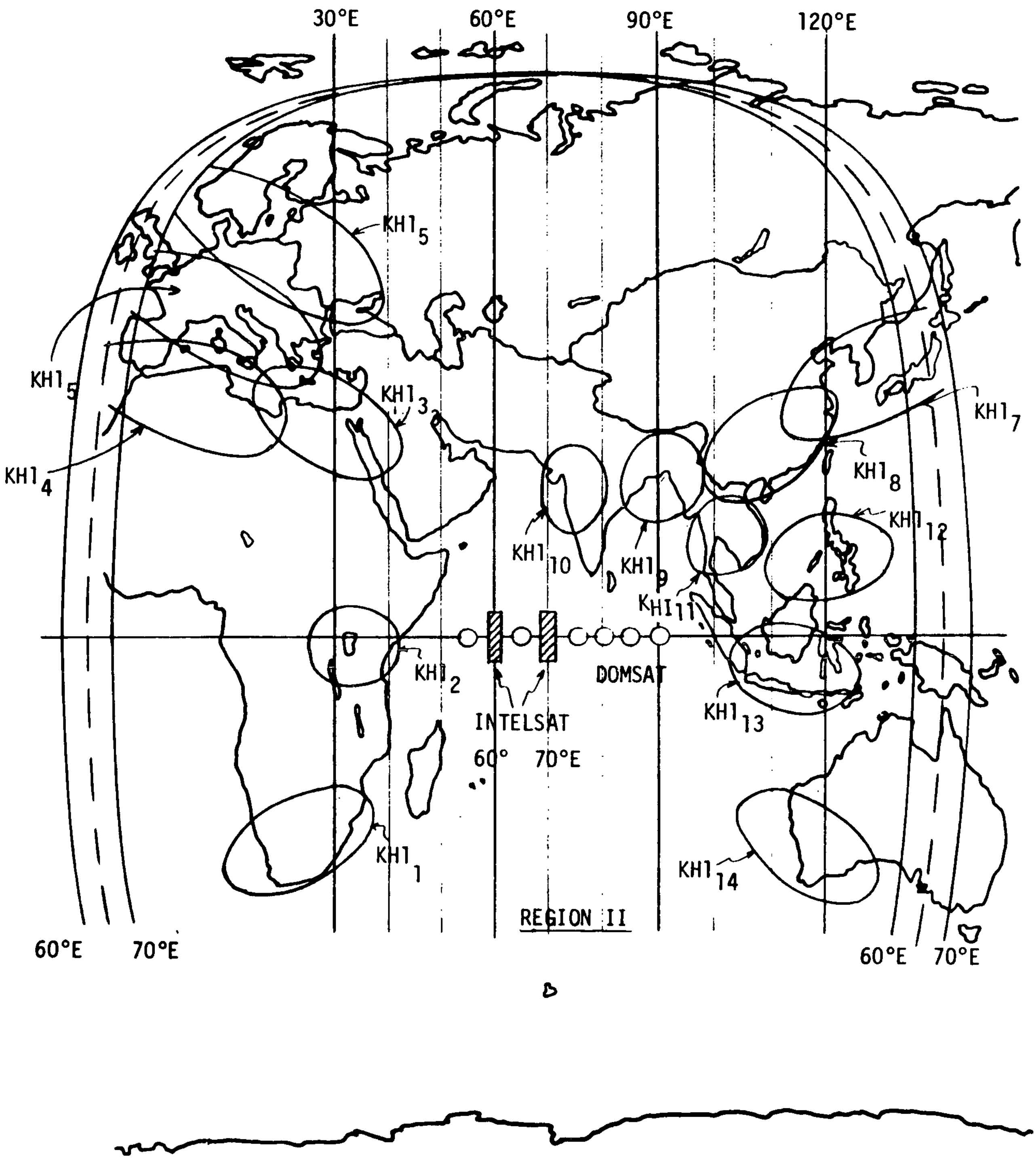


Figure 4.4-10. Intelsat Region II Platform 2° Spot Beam Pattern Layout

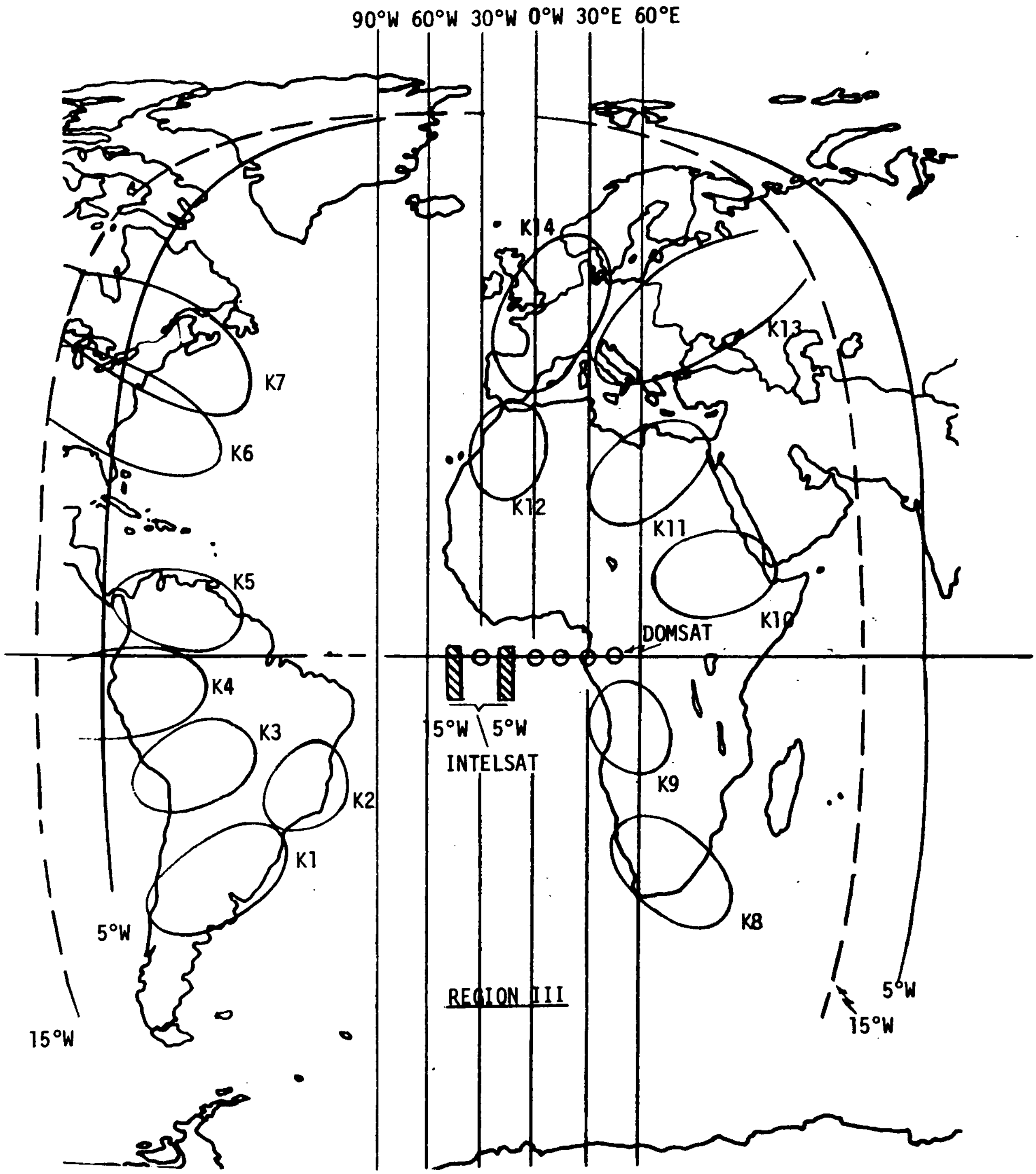


Figure 4.4-11. Intelsat Region III Platform 2° Spot Beam Pattern Layout

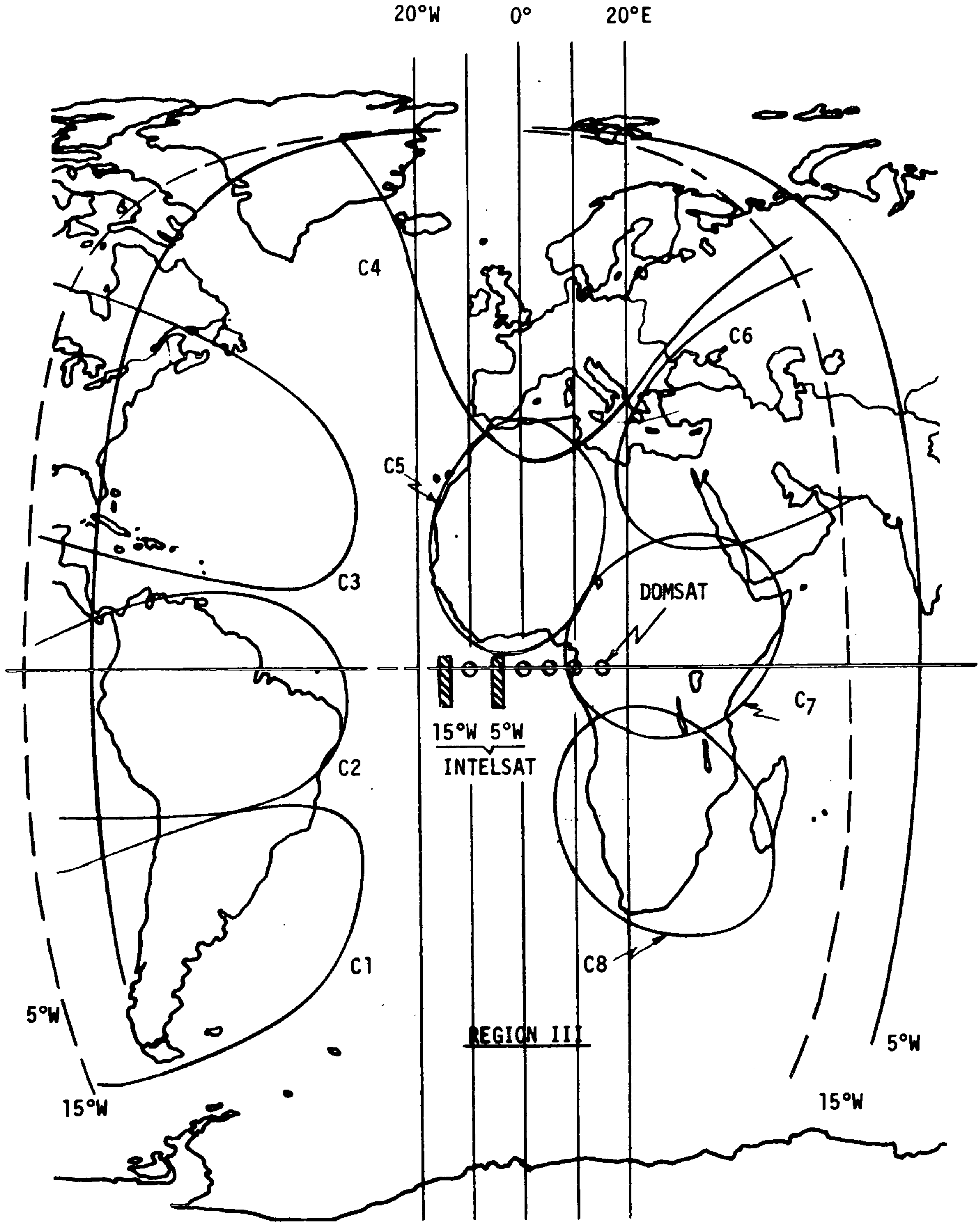


Figure 4.4-12. Intelsat Region III Platform C-Band 5° Beam Pattern Layout

Platform Electrical Configuration

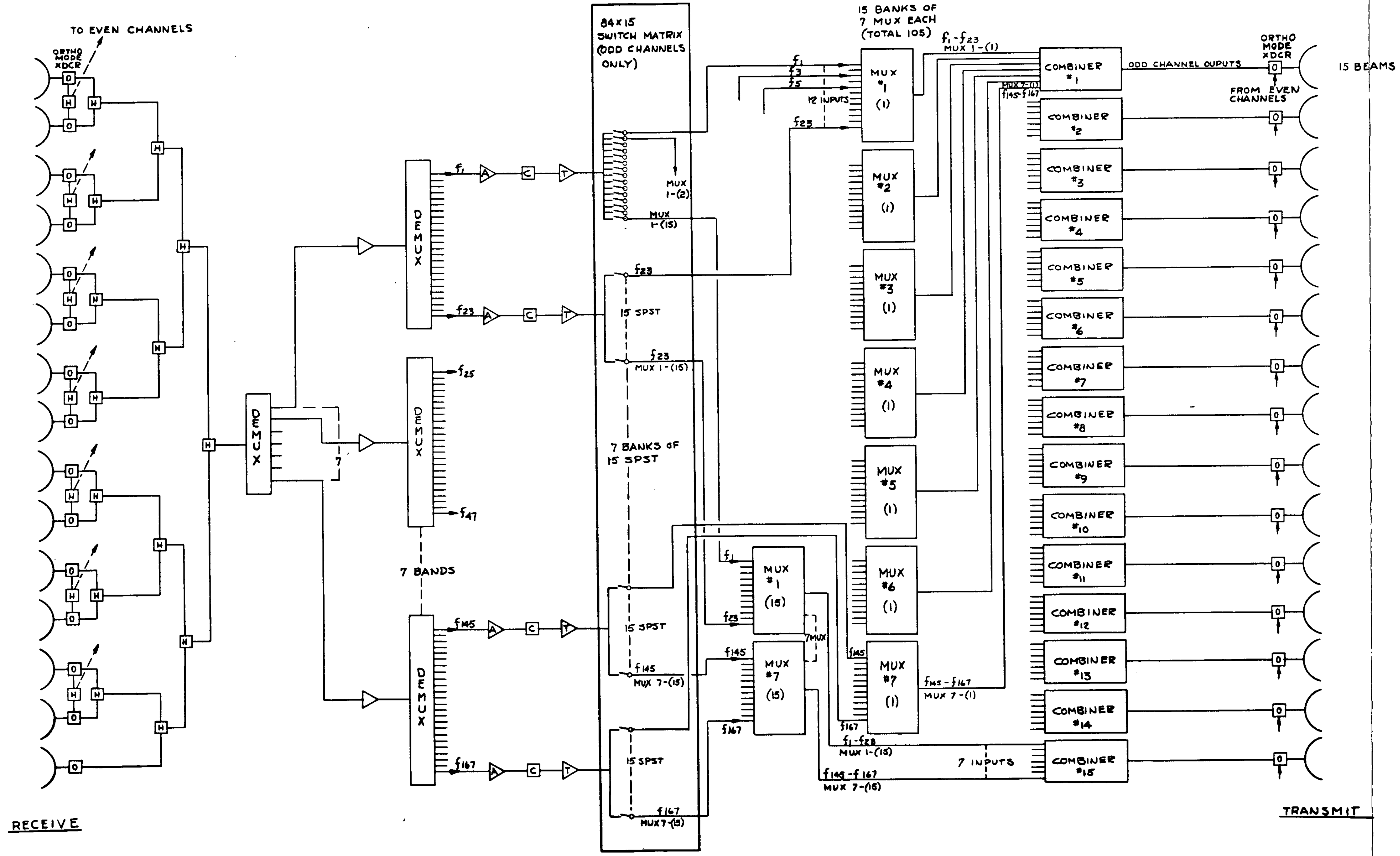
In order to provide the necessary flexibility for the traffic distribution, a quasi-TDMA system similar to the baseline traffic model platform is used on all bands. In both C- and K_{LO}-band it is essentially the same as the baseline platform equipment. At K_{HI}-band, a much more complex system evolves because of the use of orthogonal polarization and because of the large number of antenna feeds required in some Domsat regions. A switching matrix of 84 by 16 is required in the maximum case. Figure 4.4-13 is a block schematic of the arrangement for one set of polarized channels for an 84 x 15 case. A configuration for 15 antenna feeds (beams) is shown. The major differences between this and the baseline configuration is in the number of beams. This requires a series of RF hybrids on the receiver input to multiplex the 15 antenna feed outputs for the 84 odd channels.

The demultiplexing is similar to baseline with seven frequency band demultiplexers, each separating 12 individual transponder frequency channels. After amplification and conversion to the down transmit frequency and RF power amplification to one watt, each frequency is fed through a single pole, 15-position switch (PIN diodes). Each of the 15 switch outputs feeds one of the seven band multiplexers for each of 15 antenna feeds. Each group of seven multiplexers then feeds a combiner for each one of the 15 antenna feeds. Each combiner feeds through an orthomode transducer to the antenna feed. In a like manner, the even channels are fed to the antennas through the orthomode transducer. The result is the availability of two sets of 84 channels (each orthogonally polarized to the other) to each of the 15 antenna feeds. Complete flexibility is accomplished. Control of the switching matrix is accomplished by commands from ground through the TT&C link.

Transponder Mechanical Configuration

Only the maximum size K_{HI} band mechanical configuration is discussed in detail. The other systems, K_{LO}- and C-band, are essentially the same as in the baseline platform. The basic sizes of the baseline RF components at K_{HI}-band were used in this new configuration. A unique concept that utilized a building block configuration of input multiplexers, RF transponders, switching matrices, output multiplexers and combiners was postulated. A very compact arrangement resulted as shown in Figures 4.4-14 and 4.4-15. It was necessary to extend the length to 36 inches to accommodate the 15 sets of combiners, multiplexers, and switching levels. No problem is found in placing this in the modularized platform structure. Multiplexers and combiners were configured in a modular manner that allowed their interconnection in an integrated manner. The hybrids and orthomode transducers are located at the antenna assemblies. Thus, there is a total of 16 waveguide RF outputs and one RF input for the maximum box.

If less antenna feeds are necessary, the package is reduced in the 36-inch dimension by reducing the level of combiners, output multiplexers, and switching levels. Each level is 1-1/2 inches deep.



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FOLDOUT FRAME 3

Figure 4.4-13 SD 73-SA-0036-5 4-97, 4-98

SCALE NONE	DR. FENTRESS	SPACE DIVISION
	DATE 3-20-73	NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
	MODEL	

SCHEMATIC - FUNCTIONAL CHANNEL ASSIGNMENT, GEOSYNCHRONOUS PLATFORM DEFINITION STUDY — 2339-23

FOLDOUT FRAME 1

FOLDOUT FRAME 2

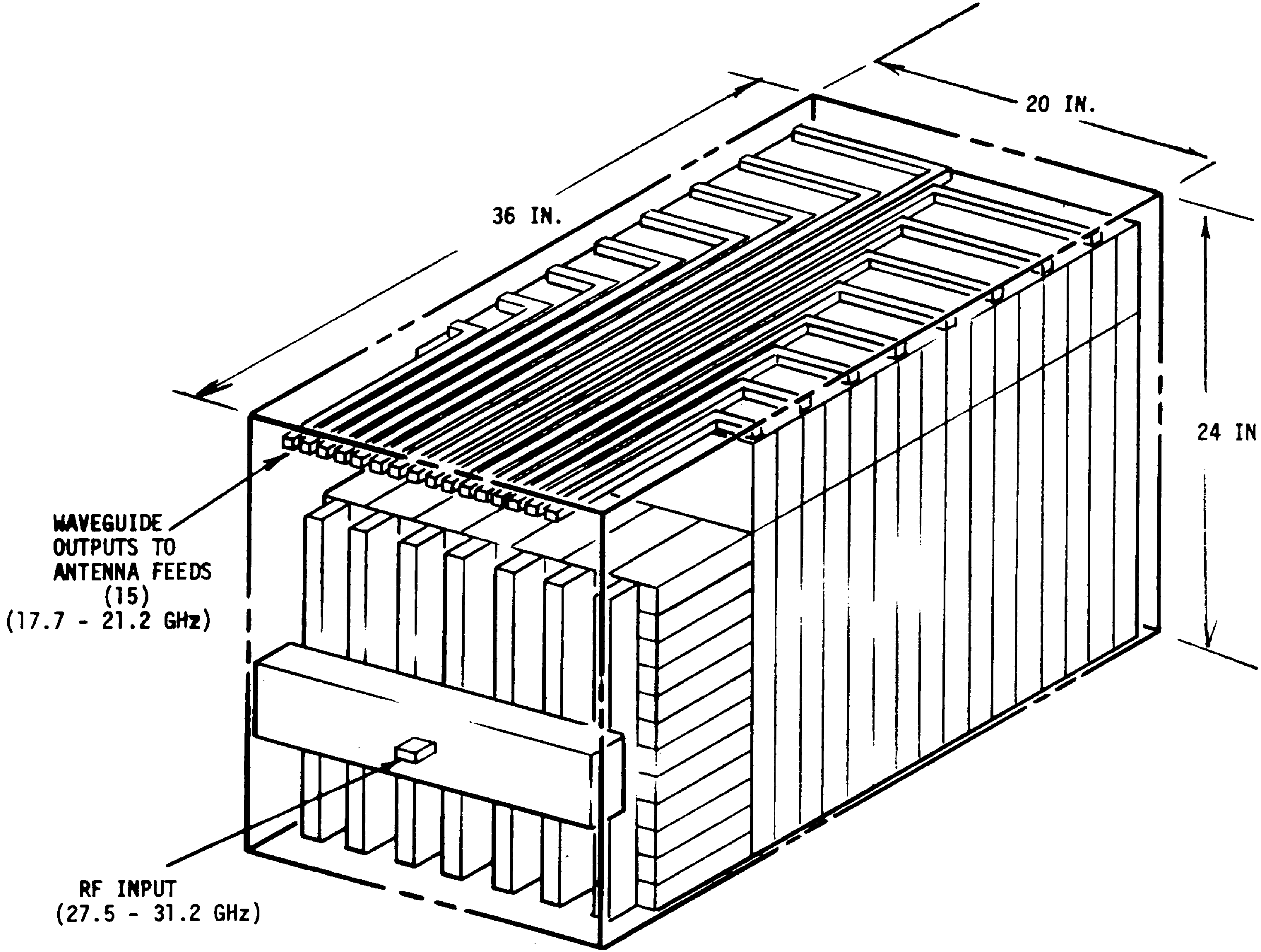

Figure 4.4-14. K_{HI}-Band 84-Channel, 16-Output Transponder/Switching Assembly Outline Drawing

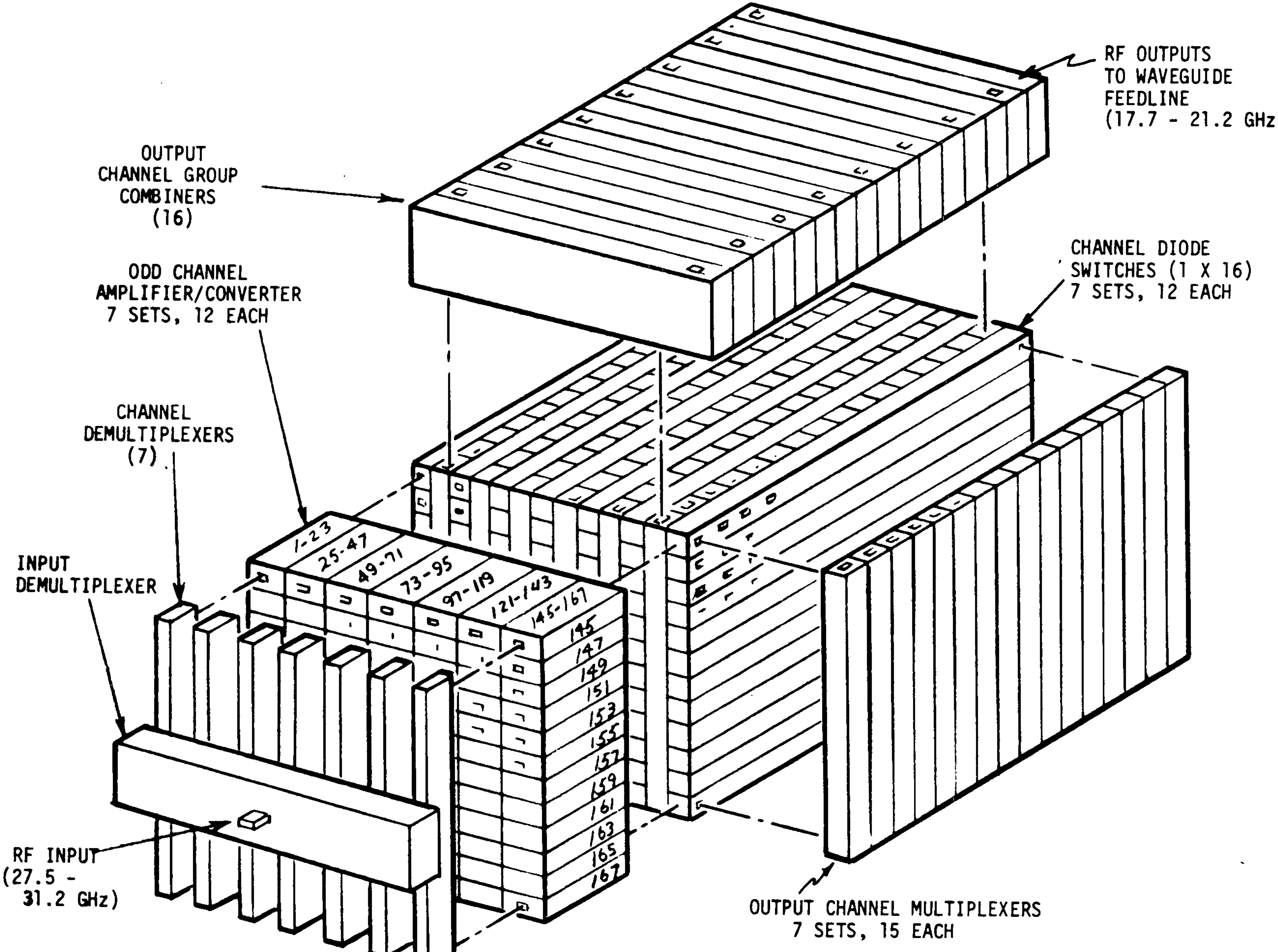

Figure 4.4-15. K_{HI}-Band Odd Channel Transponder/Switching Assembly Package
84-Channel, 16-Output System

Overall Mechanical Configuration

The basic arrangement is similar to the baseline platform. An equipment ring is used to hold the transponder packages similar to those just described. Figure 4.4-16 shows the layout of each Domsat and Intelsat platform. Four module positions hold the maximum channel capability. Where the 36-inch long K_{HI}-band units are used, alternate slots are utilized to avoid interference.

Figure 4.4-16 also shows the modular arrangements of each antenna set. Region IV required a unique configuration to accommodate the C-band shaped antennas.

Mechanical and Electrical Platform Summary

Using the same basis for weights as the baseline equipment, a set of typical data was developed for Domsat and Intelsat weights and power requirements. Tables 4.4-6 and 4.4-7 show these weights for Region IV Domsat and Intelsat platforms. The power requirements will be approximately the same for all regional platforms of the same type. Weights will vary in accordance with the number of antennas. Region IV Domsat platform carriers the largest antenna array, a total of 16. Three are shaped beam C-band antennas and the rest are parabolic antennas, 8 for K_{HI}-band, 3 for K_{LO}-band and 2 for C-band, for a total of 196 pounds of antennas. This brings the total weight for the maximum platform to 4005 pounds. Figure 4.4-16 shows the antenna arrangements for each region.

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Table 4.4-6. Typical Regional Platform Summary - Domsat Region IV

	Item	Weight (lb) Subtotal	Weight (lb) Total	Power (watts) Subtotal	Power (watts) Total
Operations Equipment	Structure		550		
	Primary	100			
	Secondary	300			
	Thermal protection	150			
	Mission equipment		1466		1385
	Transponders				
	C-band (1)	130		300	
	K_{LO}-band (1)	300		125	
	K_{HI}-band (2)	840		960	
	Antennas	66			
	C-band (2)	20			
	K_{LO}-band (3)	30			
	K_{HI}-band (8)	80			
Common Support Module	Structure		500		
	Subsystems		1489		218
	Totals		4005		1603

Table 4.4-7. Typical Regional Data Relay Platform Summary
Intelsat - Region III

	Item	Weight (lb) Subtotal	Weight (lb) Total	Power (watts) Subtotal	Power (watts) Total
Operations Equipment	Structure		540		
	Primary	100			
	Secondary	300			
	Thermal protection	140			
	Mission equipment		1080		1260
	Transponders				
	C-band (1)	130		300	
	K_{HI}-band (2)	840		960	
	Antennas				
	C-band (4)	40			
	K_{HI}-band (7)	70			
Common Support Module	Structure		500		-
	Subsystems		1489		218
	Totals		3609		1478

IMPLEMENTATION PLAN

The new traffic model extended from 1973 through 1990. During the shuttle era, 1980-1990, some of the expendable satellites in the traffic model are replacements for satellites launched prior to 1980. Other satellites are indicative of the expansion of the data relay requirements. The guideline used for the introduction of platforms was to schedule the launch of a platform when either an initial replacement satellite was required within a global zone or the expanding data relay requirement dictated the addition of another platform.

The satellite and platform schedules for the Domsats and Intelsats are presented in Figures 4.4-17 and 4.4-18, respectively. In all cases the initial platform encompasses the data relay requirements for several years. This apparent overdesign will provide a high level of redundancy during the initial phase of platform operations.

Several "replacement" satellites are indicated in the launch schedules. These satellites were required in the traffic model to replace satellites launched earlier during the shuttle era. With the on-orbit servicing concept of platforms, the replacement satellites are not required.

The standardization of transponders on the platforms resulted in excess capacity in the last platform launched in several global regions. The equivalent 24-channel transponders are indicated in the schedules.

The Region II Intelsat platforms are 48 channels short of the traffic model requirement. A third platform could have been scheduled (with an excess capacity of the equivalent of 7 satellites) but the preferred approach was to utilize the 48-channel excess capacity of the sixth Domsat platform in Region II.

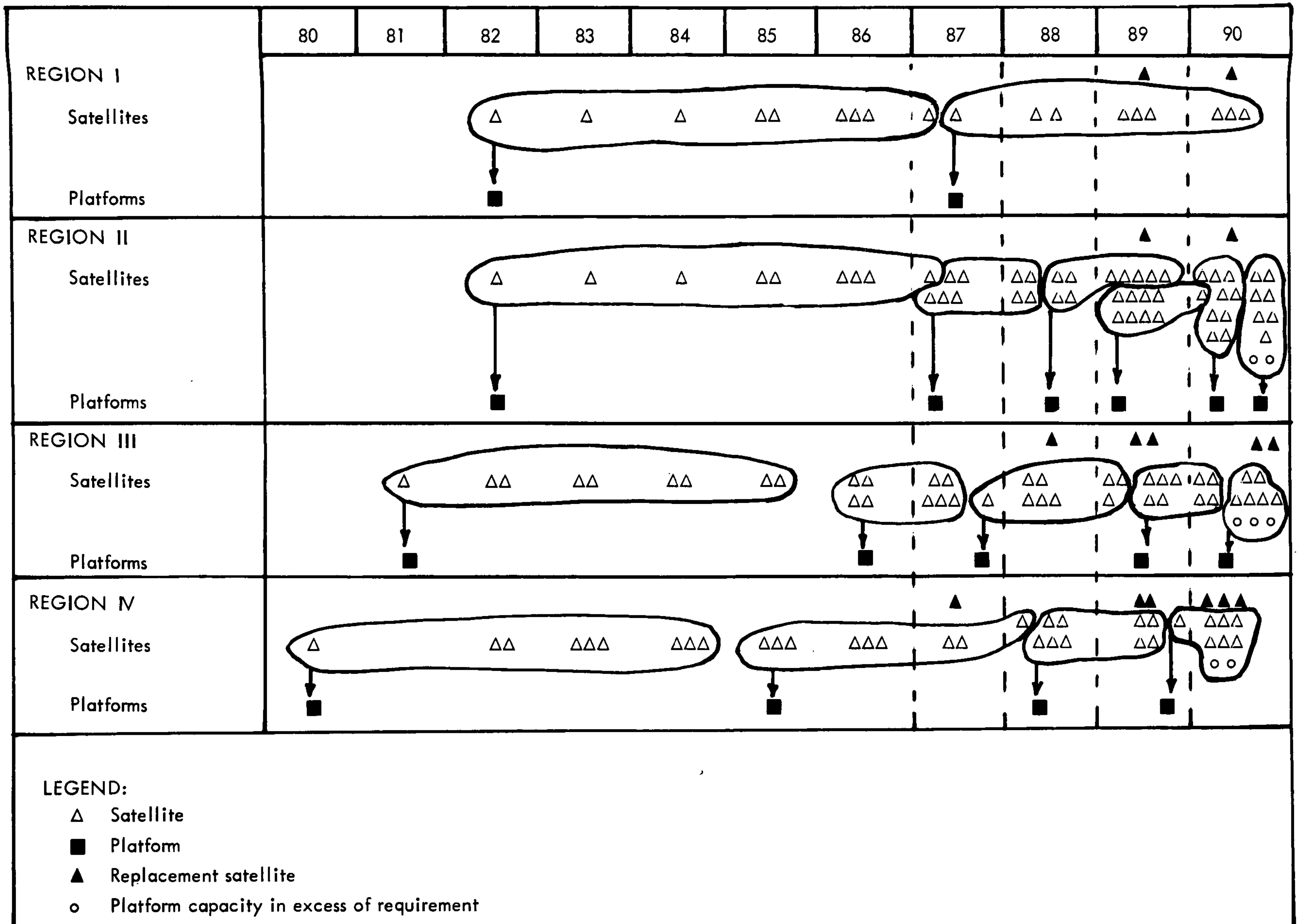

Figure 4.4-17. New Traffic Model - Domsat Implementation Plan

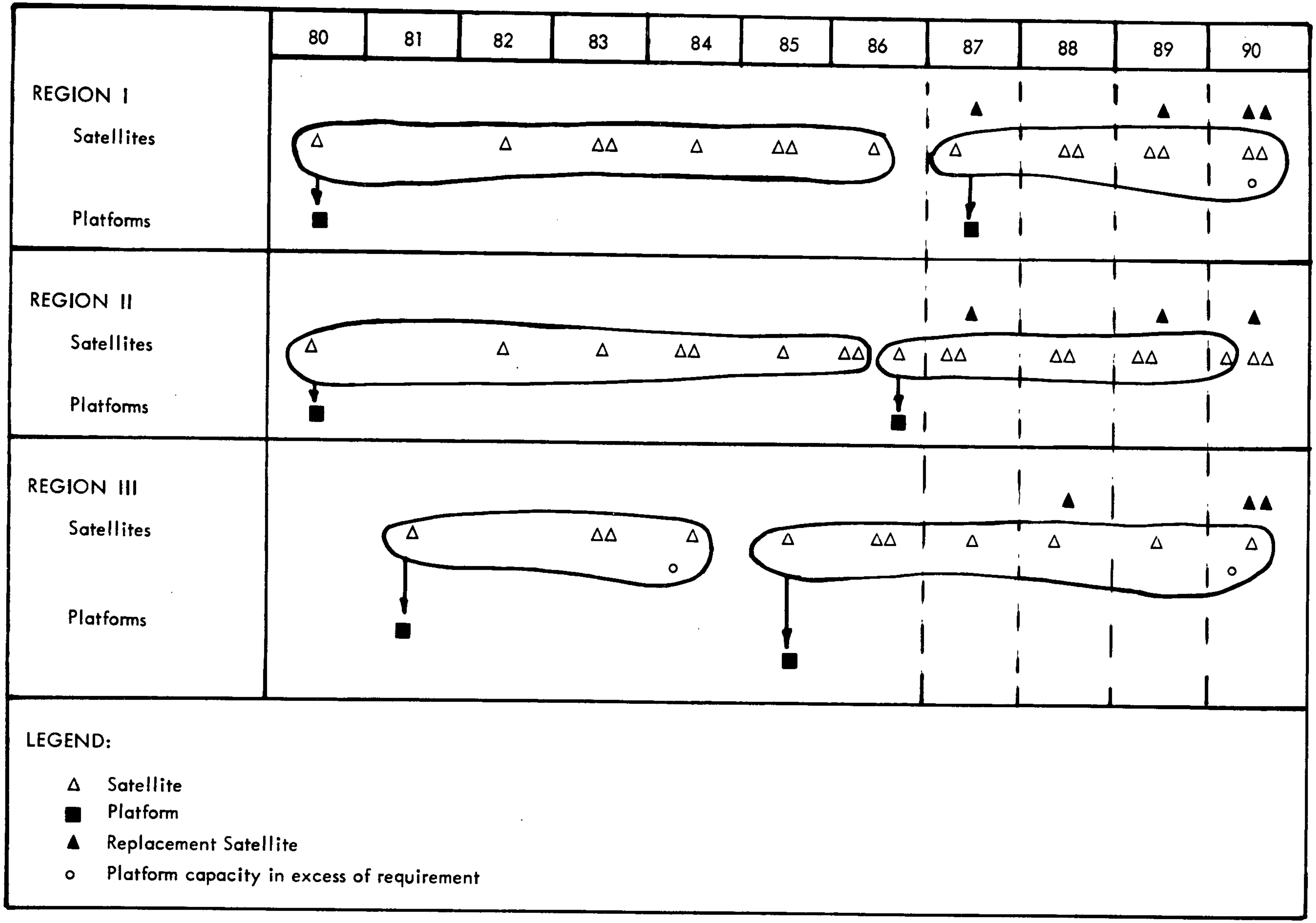

Figure 4.4-18. Intelsat Implementation Plan

4.5 TDRS PLATFORM SYNTHESIS FOR THE NEW TRAFFIC MODEL

The only difference between the two traffic models with respect to the TDRS is the addition of foreign TDRS capability. There is no technical reason for any differences between U.S. and foreign TDRS platforms. Therefore, the TDRS platform synthesis presented in Section 4.3 is also applicable to the TDRS platform for the new traffic model.

4.6 NAVIGATION AND TRAFFIC CONTROL PLATFORM SYNTHESIS

As defined in the requirements, the Nacsat system supports the traffic control function for both aircraft and ships with information that provides for (1) user to ground control communications, (2) user direct precision navigation data, and (3) ground-received automated surveillance data. Each of the five satellites in the projected regional constellations uses the same equipment and supports the functions mentioned. Two frequency bands are used for the data links. L-band (1.54 to 1.66 GHz) is used for the platform/aircraft link, and C-band (5.0 to 5.25 GHz) for the platform/ground link. Both of these bands are allocated by WARC for radio navigation use.

The design of the platform equipment must account for the characteristics of the ground station and the user aircraft or ship capabilities. As in the design for the data relay platform, the major impact on the spaceborne equipment from a weight and power consumption standpoint involves the ground and user terminal receiver capabilities. In this case, the user provides the major impact. There are many thousands of users and few space platforms. From an economic standpoint, the user equipment must be kept to a minimum consistent with feasibility of performance. Aircraft equipment is assumed to include an L-band system that uses an omni directional antenna, a transmitter capable of one-kilowatt peak power output and a receiver input noise temperature of approximately 600 K. Ground station system parameters are not limited in the same manner. The limited number of stations and their capability to support large powers and weights allows high-performance ground equipment.

Table 4.6-1 defines the functional requirements and the performance characteristics associated with these links.

The satellite transponder for the user link transmits bi-phase modulated (DPSK) digital signals that are on-board generated pseudo random noise (PN) coded signals utilized by the aircraft for range determination. Each of the platforms sends in the regional constellations its specific ephemeris data along with the PN SIGNAL. These transmissions are sent in short bursts and are sequentially timed for each of the platforms so that the user may identify the specific platform and its position. A 20 MHz bandwidth is needed to accommodate the data rate necessary for the PN code. Voice and digital data communications are interleaved with the navigation data burst.

Platform equipment consists of a transponder at L-band for the user link, processing equipment, frequency conversion equipment (L- to C-band and C- to L-band), and a C-band transponder for the ground station link.

The major weight and power driver is the L-band equipment. Based on the user receiving characteristics, an RF power output of 1000 watts with a 26 dB gain, 10-degree beamwidth antenna is necessary for transmission with adequate margin. A 10-degree beamwidth antenna provides sufficient coverage from each

Table 4.6-1. Nacsat Terminal Characteristics

Function	Link	Terminal	Frequency (MHz)	Bandwidth (MHz)	RF Power Output (watts)
Navigation	Satellite-to-aircraft	Satellite	L-band (1535 to 1558)	20	1000
Communication (digital and voice)	Satellite-to-aircraft	Satellite	L-band (1535 to 1558)	20	
Surveillance	Aircraft-to-Satellite	Aircraft	L-band (1636 to 1660)	20	1000
Communication (digital and voice)	Aircraft-to-Satellite	Aircraft	L-band (1636 to 1660)	20	
Surveillance	Satellite-to-ground	Satellite	C-band (5000 to 5250)	20	2 (peak)
Communication (digital and voice)	Satellite-to-ground	Satellite	C-band (5000 to 5250)	20	

satellite to enable users to "see" at least four satellites simultaneously. This provides the necessary number of navigation fixes. A hybrid combination of power transistors could be used to generate this power at L-band, thus providing an all-solid-state unit. The L-band system would have the following characteristics:

Operating frequency:
Transmit, 1535 to 1558 MHz
Receive, 1636 to 1660 MHz

RF power output: 1000 watts

Receiver noise temperature: 600 K

DC power input: 1500 watts

Weight: 150 pounds

Size: 20 inches by 18 inches by 20 inches

Antenna characteristics:
Parabolic dish, 4.5-foot diameter
Beamwidth (3 dB), 10 degrees
Gain, 26 dB
Weight, 15 pounds

The C-band platform equipment for the ground link can operate at much lower power levels since the ground station receiving system utilizes a high-gain antenna (37 dB compared to 0 dB for the aircraft) and low-noise receiver equipment. The RF power output to ground must be approximately 2 watts with an on-board antenna gain of 26 dB. This power level can easily be generated with a single solid-state device at 5 GHz. The C-band equipment characteristics are as follows:

Operating frequency: Transmit and receive, 5.0 to 5.25 GHz
RF power output: 2.0 watts
Receiver noise temperature: 600 K
DC power input: 25.0 watts
Weight: 35 pounds
Size: 12 inches by 12 inches by 8 inches

Antenna characteristics:
Parabolic dish, 1.5-foot diameter
Beamwidth (3 dB), 10 degrees
Gain, 26 dB
Weight, 10 pounds

The two other sets of equipment, i.e., the processing equipment for generating the ranging code and the conversion equipment for translating the RF frequencies from L- to C-band and C- to L-band, would be all solid-state and require less than 100 watts of dc power. Each of these equipment sets could be housed in a package whose dimensions are 10 inches by 8 inches by 6 inches and weighs 30 pounds.

Figure 4.6-1 describes the physical layout of the navigation and traffic control platform. Table 4.6-2 lists the physical and electrical characteristics of this platform.

Navigation and traffic control space systems are presently in formative stages. The data generated herein are based on the early concepts presently being postulated. Detailed system characteristics are intended to be representative of a possible system.

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Table 4.6-2. Navigation and Traffic Control Platform Summary

	Item	Weight (lb)		Power (watts)	
		Subtotal	Total	Subtotal	Total
Operations Equipment	Structure		570		
	Primary	100			
	Secondary	300			
	Thermal protection	170			
	Mission equipment		240		1625
	Transponders				
	L-band	150		1500	
	C-band	35		25	
	Auxiliary	30		100	
	Antennas				
	L-band	15			
	C-band	10			
Common Support Module	Structure		500		
	Subsystems		1489		218
	Total		2799		1843

5.0 OBSERVATIONAL PLATFORMS

5.1 OPERATIONS OBJECTIVES

Defined objectives for the class of observational platforms were derived from the baseline satellite inventory, and were further defined in the study in terms of scientific objectives and applications benefits. A discussion of the selection methodology and of unique geosynchronous benefits is contained in Volume IV.

The operational objectives are encompassed within the disciplinary areas of earth resources, earth physics, meteorology, solar and stellar astronomy, plasma and magnetospheric physics, high-energy physics, and radio astronomy. Twenty-two candidate payloads were selected to meet these objectives, after evaluation of their compatibility with geosynchronous orbits, and their ability to obtain benefits by operating in such orbits. These payloads are identified in Table 5.1-1.

In general, these payloads are for observing natural phenomena on earth, in the magnetosphere, or near stellar space, and benefit by their ability to observe the same, entire, earth hemisphere at all times, or to reduce certain near earth influences. More complete descriptions of the observations and geosynchronous payloads are also contained in Volume IV.

Table 5.1-1. Payloads Benefiting from Geosynchronous Orbit

Payload No.	Description
1	Synchronous earth observations
2	Synchronous meteorology
3	Synchronous earth physics
4	Synchronous earth resources
5	Solar orbiting pair (A)
6	Optical interferometer
7	Photoheliography
8	Coronagraph
9	Narrow-field ultraviolet
10	Low-energy stellar X-ray telescope
11	X-ray spectrometry/polarimetry
12	X-ray low-background stellar astronomy
13	Cometary physics
14	Meteoroids
15	Atmospheric/magnetospheric science
16	Perturbations - wake
17	VLF wave - particle interactions
18	Electron/ion beam
19	Nucleonic anti-matter
20	Extra-heavy nuclei
21	Solar X-ray telescope
22	Explorer astronomy

5.2 PAYLOADS INTEGRATION ANALYSIS

In order to prepare design concepts of platforms to accommodate the selected payloads, an integration analysis was performed. This consisted of identifying and evaluating mission equipment, defining platform criteria, and tabulating payload equipment groupings into selected concepts which would utilize the common support module and achieve defined observational objectives.

MISSION EQUIPMENT DEFINITION

A search was made of the NASA Blue Book and Aerospace Fleet analysis (References 5-1 and 5-2) and other sources for equipment defined in equivalent or comparable payload groups. Initially, 78 primary equipment items were identified, plus the associated support electronics, antennas, etc. A detailed review of the equipments indicated some redundancies of equivalent items between experiments which allowed the listing to be reduced to a final total of 65 items. The primary equipment items are listed in Table 5.2-1 in matrix with the 22 selected payloads which require the use of the mission equipment.

MISSION EQUIPMENT CHARACTERISTICS

Since the primary equipment items were selected from various sources, and frequently represented low earth orbit payloads, each item was reviewed and its characteristics revised as necessary to reflect differences in performance at geosynchronous orbit (GSO). The revised data reflect the application of scaling factors which were applied to size, weight, etc. Considerations in deriving scaling factors include the following:

1. Angular resolution to provide same spatial resolution as from low earth orbit (LEO)

2. Different range of observable parameter (e.g., low energy cosmic rays)

3. Longer observation period

4. Different background level (e.g., trapped electrons)

Scaling factors were for mass, length, height, width, volume, power, stability, pointing accuracy, attitude hold time, data rate, and environmental limits such as temperature and radiation. In general, the physical size and power characteristics tended to remain the same or be slightly greater

Table 5.2-1. Primary Equipment List

PRIMARY EQUIPMENT	PAYLOAD																					
	1 EARTH OBSERVATIONS	2 METEOROLOGY	3 EARTH PHYSICS	4 EARTH RESOURCES	5 SOLAR ORBITING PAIR	6 OPTICAL INTERFEROMETER	7 PHOTOHELIOGRAPH	8 CORONOGRAPH	9 NARROW-FIELD UV	10 LO-ENERGY STELLAR X-RAY	11 X-RAY SPECTR/PHOTOMTR	12 X-RAY LO-BKGND STELLAR	13 COMETARY PHYSICS	14 METEOROIDS	15 ATMOS/MAGNETOSPHERE	16 WAKE PERTURBATIONS	17 VLF WAVE-PARTC'L INTER.	18 ELECTRON/ION BEAM	19 NUCLEONIC ANTI-MATTER	20 EXTRA-HEAVY NUCLEI	21 SOLAR X-RAY TELESCOPE	22 EXPLORER ASTRONOMY
1. MULTISPECTRAL TV CAMERA	●	●		●																		
2. MULTISPECTRAL SCANNER	●			●																		
3. PASSIVE MICROWAVE SCANNER	●			●																		
4. MULTISPECTRAL RADIOMETER	●			●																		
5. MICROWAVE RADIOMETER	●	●	●	●																		
6. SCATTEROMETER RADIOMETER	●		●																			
7. MULTISPECTRAL SPECTROMETER	●																					
8. AERONOMY SPECTROMETER			●																			
9. SPECTRAL POLARIMETER			●																			
10. SFERICS DETECTOR			●																			
11. ABSORPTION SPECTROMETER			●																			
12. DATA COLLECTION UNIT	●	●	●	●																		
13. TELESCOPE, 1.5-METER	●																					
14. CLOUD CAMERA		●	●																			
15. HIGH-RESOLUTION IR CAMERA	●	●	●	●																		
16. X-RAY SPECTROMETER					●																	
17. PROPORTIONAL COUNTER					●																●	
18. MAGNETOGRAPH							●															
19. ULTRAVIOLET TV CAMERA									●													
20. WHITE-LIGHT TV CAMERA							●															
21. HYDROGEN ALPHA TV CAMERA							●															
22. ECHELLE SPECTROGRAPH							●															
23. SOLAR CORONAGRAPH								●														
24. XUV SPECTROHELIOGRAPH					●																	
25. SOLAR X-RAY SPECTROHELIOGRAPH					●																	
26. ELECTRIC IMAGING CAMERA					●		●	●							●							
27. PHOTOHELIOGRAPH							●	●														
28. NARROW-FIELD UV TELESCOPE									●													
29. X-RAY TELESCOPE, 0.5-METER					●																●	
30. STELLAR X-RAY TELESCOPE										●												
31. TELESCOPE ASPECT SENSOR							●	●		●	●	●									●	
32. GAMMA-RAY SPECTROMETER										●		●										
33. SPECTROMETER/POLARIMETER											●											
34. X-RAY TRANSMISSION GRATING											●											
35. X-RAY IMAGE INTENSIFIER											●											
36. X-RAY CRYSTAL SPECTROMETER					●					●											●	
37. TV FIELD CAMERA									●	●											●	
38. HIGH-DENSITY, LOW-DISPERSION SPECTROGRAPH					●				●													
39. PHOTOELECTRIC POLARIMETER											●											
40. MASS SPECTROMETER													●									
41. ELECTRON/ION MONITOR															●	●						
42. CLOSED QUADRUPOLE MASS SPECTROMETER																●						
43. LF TRANSMITTER															●		●					
44. VLF RECEIVER															●		●					●
45. PARTICLE DETECTOR					●										●		●					
46. FLUXGATE MAGNETOMETER															●		●	●				
47. ELECTRON ACCELERATOR																		●				
48. HEMISPHERICAL ANALYZER																		●				
49. TOTAL ABSORPTION SHOWER COUNTER																			●			
50. CERENKOV COUNTER																			●	●		
51. MAGNETIC SPECTROMETER																			●			
52. IR SCANNING SPECTROMETER					●																	
53. PHOTOMETRIC CLUSTER															●							
54. INTERFEROMETER/SPECTROMETER													●		●							
55. SCANNING/GRATING SPECTROMETER													●	●	●							
56. EUV SPECTROMETER					●								●		●							
57. COSMIC DUST SENSOR														●	●							
58. INFRARED INTERFEROMETER						●																
59. RADIO TELESCOPE																						●
60. GAMMA-RAY DETECTOR																						●
61. COSMIC-RAY DETECTOR																						●
62. ELF RECEIVER																						●
63. HF RECEIVER																						●
64. UHF RECEIVER																						●
65. HIGH Z DETECTOR																			●			
	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22

(i.e., factor of 1 to ∿1.5) with solar X-ray and UV sensors generally being sized at a fraction of LEO requirements (i.e., ∿0.1 to 0.5). Stability and pointing requirements tended to remain at a factor of 1 to 1.2, with hold times increasing in a few cases. Some notable exceptions in solar payload stability include factors of 30 to 300 or pointing factors of 24 to 600, which significantly relax requirements. No significant factors were found for scaling environmental limits. Table 5.2-2 identifies all equipment items affected and their scaling factor. The characteristics of each equipment item were identified, scaled as appropriate, and are listed in Table 5.2-3.

PAYLOAD GROUPING ANALYSIS

Several factors were considered in establishing the payloads that could suitably be grouped into geosynchronous platforms. These factors consisted of the following:

1. Compatibility of one payload to another
2. Savings by possible equipment commonality
3. Grouped requirements compatible with common support module capabilities
4. Sizing compatible with shuttle-tug capabilities

Commonality/Compatibility Grouping

From data in Volume IV, it was shown that payloads could have the same targets, operate on a non-interference basis, operate on a time-sharing basis, or would, in fact, interfere with each other. Based on those data, and on the primary mission equipment lists given previously (Table 5.2-1), a compatibility-commonality matrix was prepared, Table 5.2-4. From the matrix several candidate groupings can be prepared at several levels of desirability, as follows:

1. Payloads which have common targets and could make common use of a number of equipment items

Common Objective	Payloads
Earth observations	1, 2, 3, and 4
Solar astronomy	5, 7, 8, and 21
Stellar astronomy	9, 10, 11, and 12, plus 6 (no interference)
Plasma physics	17 and 18, plus 16 (no common equipment) plus 13 (no interference)
Hi-energy physics	19 and 20

2. Payloads which have common equipment--no interference

Magnetospheric physics	14 and 15

Table 5.2-2. Equipment Affected by Scaling

Primary Equipment	Scaling Factors									
	Mass	Volume	Power	Stability	Pointing Accuracy	Attitude Hold Time	Data Rate	Maximum Temperature	Minimum Temperature	Radiation
1. Multispectral TV camera	4.0	1.5	4.0	1.0	1.0	6.0	1.0	1.0	1.0	1.0
2. Multispectral scanner	3.0	2.0	4.0	1.0	1.0	6.0	1.0	1.0	1.0	1.0
3. Passive microwave scanner	4.0	5.0	4.0	1.0	1.0	6.0	1.0	1.0	1.0	-
4. Multispectral radiometer	4.0	5.0	2.0	1.0	1.0	6.0	1.0	1.0	1.0	1.0
5. Microwave radiometer	4.0	5.0	2.5	1.0	1.0	6.0	1.0	1.0	1.0	1.0
6. Scatterometer radiometer	4.0	5.0	3.0	1.0	1.0	6.0	1.0	1.0	1.0	1.0
7. Multispectral spectrometer	3.0	1.5	3.0	1.0	1.0	6.0	1.0	1.0	1.0	1.0
8. Aeronomy	2.0	2.0	4.0	1.0	1.0	6.0	1.0	1.0	1.0	1.0
9. Spectral polarimeter	2.0	1.5	2.0	1.0	1.0	6.0	1.0	1.0	1.0	1.0
10. Sferics detector	2.0	1.0	1.0	1.0	1.0	6.0	1.0	1.0	1.0	1.0
11. Absorption spectrometer	2.0	1.5	1.5	1.0	1.0	6.0	1.0	1.0	1.0	1.0
12. Data collection unit	1.0	1.0	1.0	-	-	6.0	1.0	1.0	1.0	-
13. Telescope, 1.5-meter	1.0	1.0	1.5	2.0	2.0	6.0	1.0	1.0	1.0	1.0
14. Cloud camera	1.0	1.0	1.0	-	1.0	6.0	1.0	1.0	1.0	1.0
15. High-resolution IR camera	1.5	2.5	1.0	2.0	2.0	6.0	1.0	1.0	1.0	1.0
16. X-ray spectrometer	1.0	1.0	1.0	1.0	1.0	32.0	1.0	1.0	1.0	1.0
17. Proportional counter	0.1	0.1	0.1	1.2	1.2	32.0	1.0	1.1	0.9	1.0
24. XUV spectroheliograph	0.1	0.1	0.6	30.0	24.0	1.0	1.0	1.1	0.9	<10.0
25. Solar X-ray spectroheliograph	0.1	0.1	0.2	300.0	600.0	1.2	10^{-3}	1.0	1.0	< 1.0
26. Electric imaging camera	1.0	1.0	0.6	3.0	60.0	1.0	10^{-4}	1.1	0.9	-
27. Photoheliograph	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.1	0.9	-
28. Narrow-field UV telescope	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.1	0.9	10^3
29. X-ray telescope, 0.5-meter	0.2	0.2	0.8	3.0	30.0	1.0	0.1	1.0	1.0	-
36. X-ray crystal spectrometer	0.5	0.5	0.6	3.0	4.0	1.0	0.3	1.0	1.0	-
37. TV field camera	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.1	0.9	-
38. High density, low disp. spectro.	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	10.0
41. Electron/ion monitor	1.0	1.0	1.0	1.0	1.0	1.0	0.7	1.0	1.0	-
45. Particle detector	1.0	1.0	1.0	1.0	1.0	1.0	0.3	1.0	1.0	-
49. Total absorption shower counter	1.0	1.0	1.1	-	1.0	1.0	1.0	1.0	1.0	< 1.0
50. Cerenkov counter	1.1	1.1	1.0	-	1.0	1.0	1.0	1.0	1.0	< 1.0
52. IR scanning spectrometer	1.0	1.0	1.0	1.0	1.0	1.0	0.1	1.0	1.0	-
53. Photometric cluster	1.2	1.1	1.0	-	-	1.0	1.0	-	-	-
56. EUV spectrometer	1.0	1.0	1.0	1.0	1.0	24.0	0.2	1.0	1.0	-
58. Infrared interferometer	1.0	1.0	1.0	1.0	1.0	32.0	1.0	1.0	1.0	-
59. Radio telescope	1.0	1.0	1.1	1.0	1.0	1.0	1.6	1.0	1.0	-
60. Gamma-ray detector	1.5	1.2	1.0	-	1.0	1.0	1.0	1.0	1.0	-
65. High Z detector	1.0	1.0	-	-	1.0	1.0	1.1	1.0	1.0	-

*NOTE: Equipment items not listed are all a factor of one.

Table 5.2-3. Equipment Characteristics

Primary Equipment	Characteristics*						
	Weight (lb)	Volume (cu ft)	Power (watts)	Stabiliza. (per sec)	Pointing Accuracy (±)	Data (bps)	Temperature Control (deg C)
1. Multispectral TV camera	110	1.8	200	0.2 deg	0.5 deg	10^6	+20 to +30
2. Multispectral scanner	200	8.0	130	0.05	25.0	5×10^7	0 to +40
3. Passive microwave scanner	250	2.0	175	0.05	0.1	1000	-
4. Multispectral radiometer	40	1.6	20	0.3	0.5	3000	0 to +40
5. Microwave radiometer	127	1.0	80	0.05 deg	1.0 deg	200	0 to +40
6. Scatterometer radiometer	210	3.0	150	0.05 deg	2.5 deg	3×10^4	+10 to +40
7. Multispectral spectrometer	175	6.3	170	0.05	0.1	3×10^4	0 to +40
8. Aeronomy spectrometer	101	17.0	260	0.01	1.0	4×10^4	-10 to +40
9. Spectral polarimeter	40	2.1	20	0.2	0.5	3000	+10 to +40
10. Sferics detector	22	1.8	6	0.05 deg	2.0 deg	300	+20 to +30
11. Absorption spectrometer	30	7.5	15	0.05 deg	0.05 deg	1400	0 to +40
12. Data collection unit	11	0.2	8	-	-	256/CH	-
13. Telescope, 1.5-meter	1000	-	75	2.0 arc sec	2.0 arc sec	100	+37 to +45
14. Cloud camera	14	0.5	15	-	0.1 deg	8×10^5	+8 to +38
15. High-resolution IR camera	45	2.0	24	1.0 arc min	0.1 deg	12×10^5	-20 to +30
16. X-ray spectrometer	75	2.0	150	1.0 arc sec	1.0 arc sec	300	+17 to +21
17. Proportional counter	20	8.0	22	3.0	1.0 arc min	2100	-10 to +30
18. Magnetograph	240	48.6	270	0.05	0.1 arc sec	10^7	+10 to +40
19. UV TV camera	56	2.6	100	0.05 arc sec	1.0 arc min	10^7	-10 to +40
20. White-light TV camera	56	2.6	100	0.05 arc min	1.0 arc min	10^7	-10 to +40
21. Hydrogen-alpha TV camera	56	2.6	100	0.05 arc sec	1.0 arc min	10^7	-10 to +40
22. Echelle spectrograph	90	8.0	215	0.05	0.1 arc sec	10^6	+18 to +24
23. Solar coronograph	595	-	20	1.0	15.0 "	12×10^6	+18 to +24
24. XUV spectroheliograph	90	11.2	20	3.0	1.0 arc min	2×10^6	-13 to +37
25. Solar X-ray spectroheliograph	64	10.8	47	3.0 arc sec	1.0 arc min	2900	+13 to +21
26. Electric imaging camera	10	0.3	21	40 arc sec	1.0 arc sec	50	-13 to +42
27. Photoheliograph	2640	-	40	0.01	0.1 arc sec	2000	-7 to +47
28. Narrow-field UV telescope	858	-	40	1.0	1.0 arc sec	5×10^4	+27 to +42
29. X-ray telescope, 0.5-meter	80	-	40	3.0	1.0 arc min	2000	0 to +20
30. Stellar X-ray telescope	320	-	50	1.0 arc sec	2.0 arc sec	3300	0 to +20
31. Telescope aspect sensor	160	-	19	60 arc sec	5.0 arc min	1200	-20 to +20
32. Gamma ray spectrometer	280	1.9	20	1.0 deg	1.0 deg	2500	-0 to +20
33. Spectrometer/polarimeter	60	4.8	6	1.0 arc min	5.0 arc min	100	-20 to +20
34. X-ray transmission grating	20	0.1	-	-	-	-	-
35. X-ray image intensifier	110	4.2	122	-	-	-	-20 to +20
36. X-ray crystal spectrometer	58	3.3	88	3.0 arc sec	1.0 arc min	3000	0 to +20
37. TV field camera	30	0.2	34	1.0 arc sec	1.0 arc sec	8×10^6	-13 to +32
38. High-density, low-disp. spectr.	276	28.0	50	1.0 arc sec	1.0 arc sec	4×10^6	+16 to +17
39. Photoelectric polarimeter	20	1.2	15	1.0 arc min	5.0 arc min	6100	-20 to +20
40. Mass spectrometer	31	0.2	35	-	-	10^4	-10 to +30
41. Electron/ion monitor	13	0.2	12	1.0 arc min	0.5 deg	1000	-10 to +40
42. Closed quadrupole mass spectr.	18	0.2	18	1.0 arc min	0.5 deg	1000	-30 to +60
43. LF transmitter	50	1.8	50	-	-	10	0 to +40
44. VLF receiver	73	0.6	20	-	-	2000	0 to +40
45. Particle detector	75	3.0	50	1.0 arc min	1.0 deg	12×10^3	0 to +40
46. Fluxgate magnetometer	3	0.1	20	-	5.0 deg	1000	0 to +40
47. Electron accelerator	100	21.0	130	1.0 arc min	1.0 deg	10^4	0 to +60
48. Hemispherical analyzer	8	0.4	5	1.0 arc min	1.0 deg	10^4	0 to +60
49. Total absorption shower counter	3000	15.0	44	-	1.0 deg	200	+10 to +30
50. Cerenkov counter	1020	28.0	10	-	1.0 deg	100	+10 to +30
51. Magnetic spectrometer	200	34.0	500	-	-	500	0 to +30
52. IR scanning spectrometer	65	4.3	50	1.0 arc min	0.1 deg	2000	-20 to +30
53. Photometric cluster	38	2.2	25	-	-	8000	-
54. Interferometer/spectrometer	234	15.7	457	0.01 deg	2.0 deg	4×10^4	0 to +30
55. Scanning/grating spectrometer	254	17.8	457	0.01 deg	2.0 deg	2×10^4	0 to +30
56. EUV spectrometer	39	1.4	16	3.0 arc min	2.0 arc min	13×10^3	0 to +30
57. Cosmic dust sensor	21	1.5	10	-	-	50	-
58. Infrared interferometer	40	3.0	50	0.05 arc sec	0.1 arc sec	2×10^4	0 to +30
59. Radio telescope	1000	-	-	0.5 deg	1.0 deg	3200	-33 to +70
60. Gamma-ray detector	48	0.3	10	-	5.0 deg	4000	+5 to +30
61. Cosmic-ray detector	130	25.0	15	-	5.0 deg	1200	+5 to +40
62. ELF receiver	12	0.5	20	-	-	2000	0 to +50
63. HF receiver	12	0.5	20	-	-	2000	0 to +50
64. UHF receiver	12	0.5	20	-	-	2000	0 to +50
65. High Z detector	304	4.5	-	-	1.0 deg	1000	+2 to +25

*Includes support equipment weight and power, and volume of internally installed primary equipment and support electronics.

Table 5.2-4. Science Payloads Compatibility

Payload	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22
1. Earth observations	-	1	1	1	X	X	X	X	X	X	X	X	3	2	2	3	3	3	3	3	X	X
2. Meteorology		-	1	1	X	X	X	X	X	X	X	X	3	2	2	3	3	3	3	3	X	X
3. Earth physics			-	1	X	X	X	X	X	X	X	X	2	2	1	3	3	3	3	3	X	X
4. Earth resources				-	X	X	X	X	X	X	X	X	3	2	2	3	3	3	3	3	X	X
5. Solar orbit observatory					-	3	1	1	3	3	3	3	X	2	3	X	X	X	3	3	1	X
6. Optical interferometer						-	3	3	1	2	2	2	X	2	3	X	X	X	2	2	3	X
7. Photoheliograph							-	1	3	3	3	3	X	2	3	X	X	X	X	X	1	X
8. Coronograph								-	3	3	3	3	X	2	3	X	X	X	X	X	1	X
9. Narrow-field ultraviolet									-	1	1	1	X	2	3	X	X	X	2	2	3	X
10. Low-energy stellar X-ray										-	1	1	X	2	3	X	X	X	2	2	3	X
11. X-ray spectro-polarimetry											-	1	X	2	3	X	X	X	2	2	3	X
12. X-ray low-background stellar												-	X	2	3	X	X	X	2	2	3	X
13. Cometary physics													-	2	X	1	1	1	X	X	X	X
14. Meteoroids														-	2	2	2	2	2	2	2	2
15. Atmosphere/magnetosphere															-	X	X	X	2	2	3	X
16. Wake perturbations																-	1	1	X	X	X	X
17. VLF wave-particle interaction																	-	1	X	X	X	X
18. Electron-ion beam																		-	X	X	X	X
19. Nucleonic anti-matter																			-	1	3	2
20. Extra-heavy nuclei																				-	3	2
21. Solar X-ray telescope																					-	X
22. Explorer																						-

1 = Common target or parameter
2 = No interference
3 = Time-sharing required
X = Incompatible
O = Use of same equipment

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North American Rockwell

System Requirements Versus Capabilities

Common support module capabilities were selected on the basis of a range of satellite requirements then sized to meet a range of potential platform groupings. In order to assess the payloads for compatibility with common support modules, and to define any potential grouping constraints, the weight, power, number of modules, and data rates were summed for all 22 of the baseline payloads. These characteristics identify potential requirements for platforms in the subsequent grouping concepts. The baseline payloads are shown in Table 5.2-5. None of the individual payloads exceeds the nearly two kilowatts of power previously identified as a common support module plateau, and various payloads can be combined even with straight summing of power levels without exceeding this constraint. The common support module structural tier will contain 12 components and again, various combinations can be accommodated within one, two, or three tiers. The combined data rates of two payloads slightly exceed data rate targets of the defined common support module; however, all instruments are not required to operate at the same time. Therefore, the common support module can accommodate the operational data rates of these combined payloads.

The other subsystems are sized to handle performance requirements of the platforms and, therefore, do not present limitations on the mission equipment groupings. While some optical sensors require greater stability and pointing accuracy than the common support module capability of one arc second per second and 10 arc seconds, respectively, these requirements are normally met by separately gimbaling the optical sensor itself and utilizing the sensor as the information source for controlling the servo loop.

One significant exception involved the radio astronomy platform. Definition of this payload included a large (10 kilometers) rhombic antenna which would be deployed and stabilized in a plane normal to the spacecraft cylindrical axis by subsatellites, while precessing so as to sweep for radio sources in several low frequency ranges. No reasonable extension of common support module RCS or stabilization and control subsystems was adequate for this concept. Consequently, this payload was not considered to be suitable for grouping onto a platform, and for purposes of this study was considered to be a unique satellite design. No further evaluation of this payload was conducted in this study.

In summary, with the exception of radio astronomy, no constraint on payload grouping was imposed by common support module capabilities.

Platform Sizing Constraints

A final constraint on grouping of platforms was logistics system capabilities. Since a shuttle-tug combination leaves less than 24 feet of length available for a payload, the payload candidates were screened for potential sizing constraints. Several telescopes are included in the mission equipment. Since the longest was only slightly over 16 feet, no single item would be excluded. Potential problems in grouping telescopes and large antennas, etc., were deferred for consideration in the design integration.

Table 5.2-5. Baseline Payload Definition

Discipline	Payload	Mission Equipment			
		Weight (lb)	Power (watts)	Number of Replaceable Modules	Data (bps)
Earth Observations	1. Synchronous earth observations	2170	1030	11	5×10^7
	2. Synchronous meteorology	300	330	5	1×10^6
	3. Synchronous earth physics	600	580	11	2×10^6
	4. Synchronous earth resources	780	640	8	5×10^7
Solar Astronomy	5. Solar orbiting pair	850	550	12	6×10^6
Stellar Astronomy	6. Optical interferometer	40	50	2	50
Solar Astronomy	7. Photoheliograph	3250	770	8	3×10^7
	8. Coronograph	3400	100	2	12×10^6
Stellar Astronomy	9. Narrow-field ultraviolet	1220	224	4	2×10^7
	10. Low-energy stellar X-ray	850	210	5	8×10^6
	11. X-ray spectral photometry	370	160	4	7400
	12. X-ray low-background stellar	440	40	2	3700
Plasma and Magnetospheric Physics	13. Cometary physics	560	970	8	8×10^4
	14. Meteoroids	280	470	3	2×10^4
	15. Atmosphere/magnetosphere	810	1140	14	10^5
	16. Wake-perturbations	30	30	2	2000
	17. VLF wave-particle interactions	200	140	4	15×10^3
	18. Electron-ion beam	110	160	3	21×10^3
High-Energy Physics	19. Nucleonic anti-matter	4520	550	4	1800
	20. Extra-heavy nuclei	1020	10	1	100
Solar Astronomy	21. Solar X-ray telescope	590	210	4	8×10^6
High-Energy Physics	22. Explorer	1290	110	7	4×10^4

A review of payload weight capabilities was conducted for both single and dual tug launches. In comparing the relative benefits and penalties, it became apparent that single tug payload operations were preferred. Two candidate groupings were evaluated further. These consisted of (1) a 4250-pound limit per platform that allows dual platform emplacement or single platform retrieval, and (2) an 8500-pound limit that restricts operation to single tug mission emplacement. The first group was selected based on the fact that some individual satellites approached the first limit (4250 pounds). Two-tug operations (payloads greater than 8500 pounds) were eliminated since the inherent incompatibility of the mission equipment associated with the various operations would prohibit grouping for efficient use of the tandem tug capacity. Therefore, 8500 pounds represented an upper limit for grouped platform weights. The mission capabilities for the various platform weight plateaus and delivery/retrieval options are given in Table 5.2-6.

Candidate Platforms

An initial grouping of equipment compatible satellite candidates was prepared from the data in Table 5.2-5 to meet the retrieval/dual delivery target weight (4250 pounds). This resulted in a total of ten potential platforms in lieu of 22 satellites. Rationale for the groupings includes an estimated 2200 pounds allowable for mission equipment out of the total 4250 pounds for mission equipment, subsystems, and structures. In some cases, no feasible combination was possible within the weight constraint; in fact, three payloads would individually exceed retrieval capabilities. One combination resulted in excessive power requirements for the common support module, which might necessitate time-sharing. Characteristics of these candidates are summarized in Table 5.2-7. A significant benefit was achieved in combining platforms in that 15 equipment items could be used for two or more objectives, thus saving weight-to-orbit, power, etc. For example, summing the weight of payloads 2, 3, 4, and 15 would give 2490 pounds versus the integrated weight of 2010. The reasons for this can be readily observed by referring back to Table 5.2-1.

The grouping for a single-tug delivery-only mission (8500 pounds) allowed 3000 pounds for structures and support systems. This grouping reduced the original 22 payloads to five candidate platforms. In several cases, the compatibility constraint was reached at much lower weights than 8500 pounds. Overall, 23 items of mission equipment were eliminated through integration of functions with benefits such as over 1000 watts of average power reduction from the sum for individual payloads. Except for average power, no platform was unreasonable with respect to compatibility, commonality, constraints, or requirements. Characteristics are summarized in Table 5.2-8.

Summary of Platform Selection Analysis

The selected platform groupings are based on the 8500-pound, delivery-only candidates. Rationale for the selection is based on several factors.

1. The general trend for platforms is to maximize grouping while retaining functional objectives. This is best achieved in the five-platform group.

Table 5.2-6. Payload Delivery/Retrieval Options - Weight Only

Platform Weight (lb)	Number of Platforms		
	Deliver Only	Retrieve Only	Deliver + Retrieve
Single Tug			
3300	2	1	1 + 1
3301 to 4250 *	2	1	-
4251 to 4600	1	1	-
4601 to 8500 *	1	-	-
Dual Tug			
3300	7	4	3 + 3
3301 to 4600	5	3	2 + 2
4601 to 5000	4	3	1 + 1
5001 to 12,000	2	1	-
12,001 to 15,000	1	1	-
15,001 to 24,000	1	-	-

* Candidate Groups

Table 5.2-7. Retrieval Weight Candidate Platforms

Candidate Platforms	Payload Numbers	Weight* (lb)	Power (watts)	Number RU's	Data Rate (bits/sec)
1. Earth Observations	1	2200	1030	11	5×10^7
2. Meteorology and Earth Physics	2, 3, 4, and 15	2010	2240	30	4×10^7
3. Solar/X-Ray Astronomy	5 and 21	1360	660	14	14×10^6
4. Photoheliograph	7	3250	765	8	3×10^7
5. Coronograph	8	3400	100	2	12×10^6
6. Narrow Field UV	9	1220	224	4	2×10^7
7. Stellar Astronomy	6, 10, 11, 12, and 14	1370	871	28	8×10^6
8. Plasma Physics	13, 16, 17, and 18	900	1270	14	14×10^4
9. High-Energy Physics	19	4520	550	4	2000
10. Magnetospheric Physics	20 and 22	1200	100	4	4×10^4

* 2200 Pounds Allowable

Table 5.2-8. Delivery-Only Weight Candidate Platforms

Candidate Platforms	Payload Numbers	Weight* (lb)	Power (watts)	Number RU's	Data Rate (bits/sec)
1. Earth Sciences and Meteorology	1, 2, 3, 4, and 15	3340	2700	31	6×10^7
2. Solar Astronomy	5, 7, 8, and 21	5530	1400	20	4×10^7
3. Stellar/X-Ray Astronomy	6, 9, 10, 11, and 12	2320	630	14	2×10^7
4. Plasma Physics	13, 14, 16, 17, and 18	920	1280	17	1×10^5
5. High-Energy and Magnetosphere Physics	19, 20 (22)	4410	580	8	5×10^4

* 5500 Pounds Allowable

2. Several of the individual payloads cannot meet the retrieval criteria. Therefore, the moderate weight platforms would not all meet desired goals.

3. The platform concept with common support modules is based on servicing not retrieval; therefore, delivery only is a more compatible objective.

4. Significant benefits (weight, power, number of instruments, etc. ≃ cost) were demonstrated by the grouping analysis, and are maximized in the five-platform group.

5. The larger weight platform group is much more homogeneous in characteristics and requirements and is thus more compatible with the logistics system, servicing concepts, and the common support module.

The weight and power for payloads and the alternate platform concepts are shown in Figures 5.2-1 and 5.2-2.

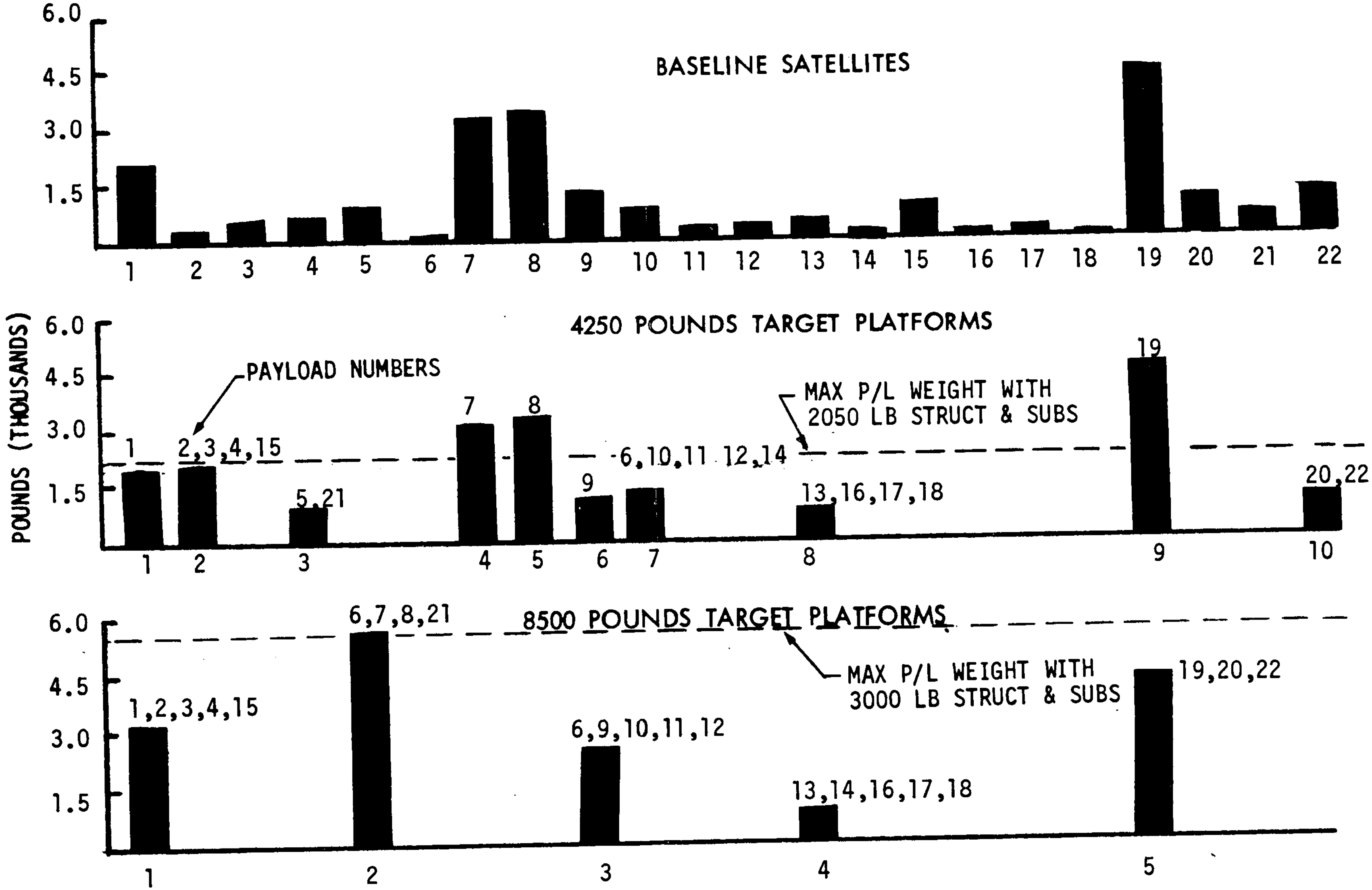

Figure 5.2-1. Observation Mission Equipment Weights

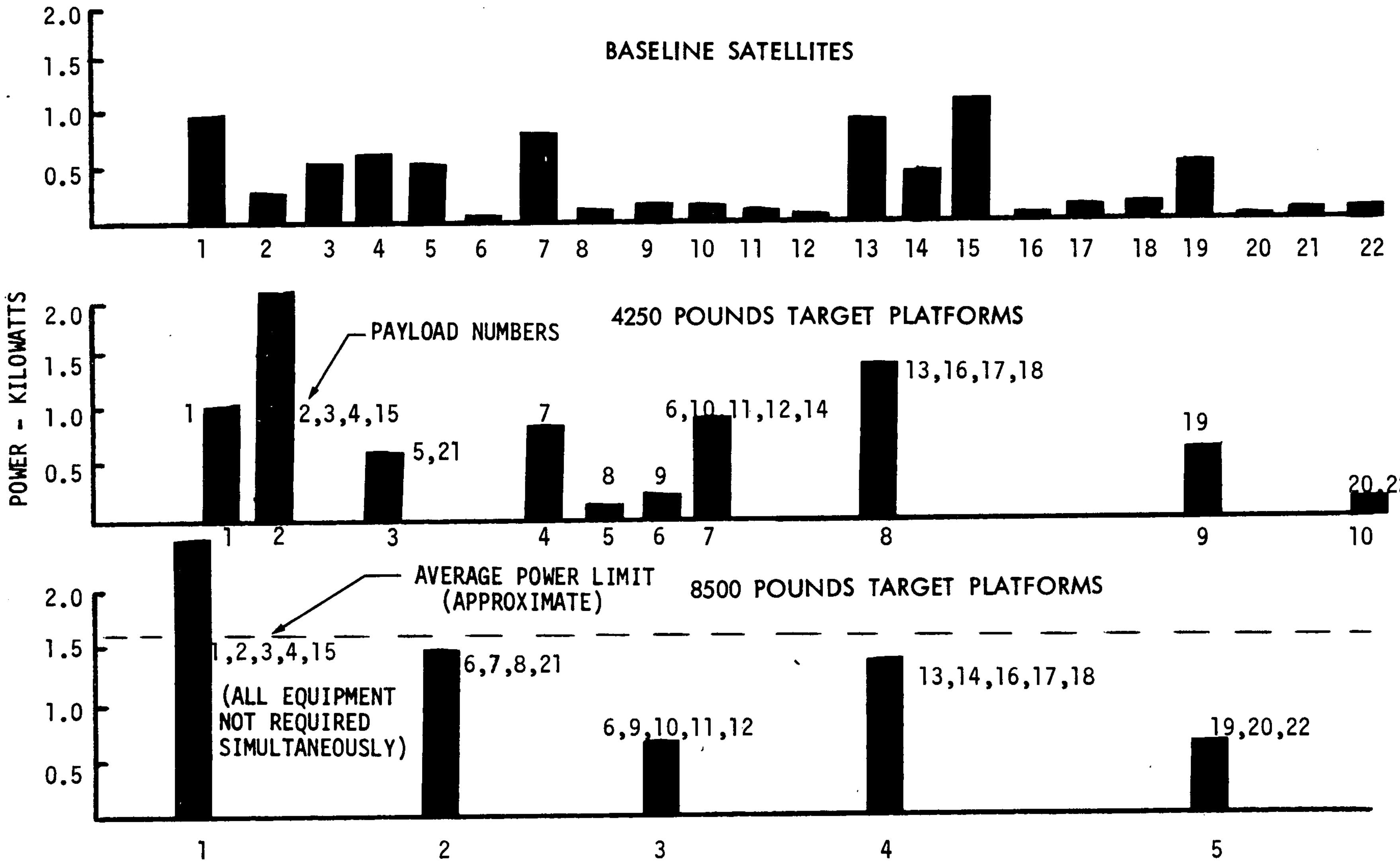

Figure 5.2-2. Observation Mission Equipment Maximum Power
(Sum of all Equipment Power Requirements)

5.3 PLATFORM DESIGN CONCEPTS

The selected five science platform groupings were individually analyzed in more depth and integrated into overall spacecraft designs. Factors considered in the design integration were:

- Platform requirements from Volume IV
- Servicing system requirements and concepts
- Common support module design concepts
- Mission equipment groupings
- Mission equipment functional arrangement for the required operational mode

Each of the five platforms is defined in detail as to its operational mode, equipment characteristics, and overall design arrangement.

EARTH OBSERVATIONS GEOSYNCHRONOUS PLATFORM

All earth resources, earth physics, atmospheric, and meteorology objectives were found to be feasibly combined into one all-purpose earth observation facility. The major element of the facility is the 1.5-meter Cassegrain-type telescope to provide the capability to meet the resolution required for imaging from geosynchronous altitude. The telescope will provide a footprint less than 100 miles and resolution in the order of 100 feet. The telescope provides a high-resolution capability for various IR and spectral optical, electronic imaging sensors used for mapping, and resource and pollution identification. Three large antennas provide the desired high resolution for microwave scanning, and provide boundary mapping, or surface and atmosphere temperatures and moisture delineation.

As an example, a passive microwave scanner described for low earth orbit, if put into synchronous orbit, would have to be about 60 feet in diameter and would require a gain of 57 dB to get similar results. (Note: 60 dB gain is the present-day state of the art.)

It would appear that a 60-foot antenna is feasible with the present-day state of the art, including the ability to obtain adequate gain. However, using active radar at synchronous altitude is prohibitive with respect to both the gain and with the amount of radiated power required. Calculations indicate that approximately 90 dB additional gain is required to overcome transmission losses. This means that a power increase of 10^9 is required for the active microwave radar system alone when flown at the synchronous altitude.

Functional Performance

Overall, the earth observation platform and polar and low-earth orbit satellites are mutually supplementary with the GSO observations having some advantages as to ground track coverage and near-constant viewing of target sites. Individually, the platform has a capability for a broad range of data collection. In addition, various earth characteristics can be measured by more than one technique, thus providing cross-correlation of data. A table of sensor functional acquisitions (Table 5.3-1) shows the range of capabilities.

Physical Description

The earth observations geosynchronous platform (reference Figure 5.3-1) consists primarily of equipment rings mounted fore and aft on the 1.5-meter telescope barrel. The aft (-Z) equipment ring is at the focal point of the telescope (behind the primary optics) where various optical sensors are coupled to the telescope light beam. Other sensors are mounted in this ring for viewing ambient magnetospheric phenomena. Two forward equipment rings house subsystems (common support module) and direct earth-viewing sensors. These rings also act as a sunshade for the telescope, which looks directly through the core section. Three large dish antennas (30, 30, and 60 feet) and a large (123 feet tip-to-tip) dipole antenna dominate the exterior appearance. In addition, several smaller sensors are externally mounted to provide for separation from the spacecraft or, as in the case of the scan-platform assembly, to enable pointing capability. A docking assembly is provided at both ends (+Z and -Z) to accommodate the servicing system for exchange of replaceable modules.

Table 5.3-2 identifies all mission equipment and its characteristics. The overall platform weight of 8496 pounds is summarized in Table 5.3-3.

SOLAR ASTRONOMY GEOSYNCHRONOUS PLATFORM

The solar astronomy platform contains four major optical assembly groupings to perform investigations and obtain solar images in X-ray astronomy and ultraviolet, photoheliograph profiles, and coronographs. The payload grouping allows multiple sensor utilization for solar phenomena at a high level of resolution.

Functional Performance

Scientific objectives were discussed briefly in Volume IV, and are covered thoroughly in NASA source data. Table 5.3-4 summarizes sensor functional performance.

Physical Description

The solar astronomy geosynchronous platform (reference Figure 5.3-2) consists of three standard equipment rings, and several optical/telescope assemblies. The aft (-Z) or common support module ring contains the docking mechanism for servicing and houses the subsystems. The mid-ring primarily contains support electronics and imaging devices for forward-mounted sensors. The forward (+Z) ring is nonstandard to accommodate unique mission equipments which may require light path acquisition or placement for exterior viewing. Forward of this ring are telescopes ranging from 0.25 meter to 1.5 meters in size.

Table 5.3-5 provides mission equipment characteristics. The overall weight of the platform (8302 pounds) is summarized in Table 5.3-6.

Table 5.3-1. Earth Observations Sensor Functional Summary

EQUIPMENT ITEM	PRIMARY FUNCTIONS																				
	LITHOLOGY	GEOLOGY	EARTH STRUCTURE	TOPOGRAPHY	WATER RUN-OFF	OCEAN POLLUTION	SNOW/ICE	VEGETATION	OCEAN COLOR/LIFE	SURFACE TEMPERATURE	RESOURCE RECOGNITION	LAND USE	MOISTURE BOUNDARIES	SOIL TEXTURE	SEA STATE	CLOUD CONTENT	AURORA/AIRGLOW	LIGHT POLARIZATION	STORM MAPPING	AIR POLLUTION	MAGNETOSPHERE WAVE/PARTICLES
MULTISPECTRAL TV	X		X	X	X	X		X	X		X	X								X	
MULTISPECTRAL IR SCANNER	X	X	X	X	X	X		X	X	X	X	X	X							X	
PASSIVE MICROWAVE SCANNER		X	X		X	X	X			X	X		X	X							
MULTISPECTRAL RADIOMETER					X	X		X	X	X		X	X	X	X						
AERONOMY SPECTROMETER																	X		X	X	
SPECTRAL POLARIMETER																X	X	X		X	
SFERICS DETECTOR																X	X		X		
ABSORPTION SPECTROMETER																	X			X	
CLOUD CAMERA (3-COLOR SCAN)																X			X		
HIGH-RESOLUTION IR CAMERA						X		X	X	X										X	
ELECTRON/ION MONITOR																					X
LF-VLF TRANSCEIVER																					X
PARTICLE DETECTOR																					X
FLUXGATE MAGNETOMETER																					X
PHOTOMETRIC CLUSTER																					X
INTERFEROMETER/SPECTROMETER																	X				X
SCANNING/GRATING SPECTROMETER																	X				
EUV SPECTROMETER																					X
COSMIC DUST SENSOR																					X
MICROWAVE RADIOMETER	X						X			X	X	X	X	X		X					
SCATTEROMETER RADIOMETER		X	X				X			X		X	X	X	X						
MULTISPECTRAL SPECTROMETER		X	X		X			X			X	X									

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Table 5.3-2 . Earth Observations Platform
Mission Equipment Characteristics

Equipment	Weight (lb)	Power (watts)	Dimensions (in.)
TELESCOPE ASSEMBLY	1000	75	70 d x 180
TELESCOPE RING--IMAGING SENSORS			
Multispectral IR scanner	125	30	19 x 22 x 24
Support electronics	75	100	12 x 12 x 24
Multispectral TV	110	200	6 x 18 x 24
Multispectral radiometer	40	20	20 d x 24
Multispectral spectrometer	175	170	20 x 18 x 30
High-resolution IR camera	45	24	12 d x 24
TELESCOPE RING--EXTERNAL SENSORS AND ELECTRONICS			
Fluxgate magnetometer	3	20	3 x 3 x 3
LF-VLF transceiver	27	70	20 x 20 x 24
Particle detector	75	50	12 x 12 x 36
Cosmic dust detector	7	-	16 x 15 d
Support electronics	14	10	19 x 14 x 6
Ion temperature and desnsity sensor	13	12	12 x 6 d
EUV spectrometer electronics	14	16	9 x 9 x 16
Scanning-grating spectrometer electronics	68	457	19 x 20 x 16
Interferometer spectrometer electronics	68	457	19 x 21 x 16
SCANNING PLATFORM			
EUV spectrometer	25	-	10 x 12 d
Scanning-grating spectrometer	186	-	24 x 40 x 46
Interferometer spectrometer	166	-	29 x 21 x 39
Photometric cluster	38	25	12 x 24 x 12
EARTH-POINTING RING			
High-resolution IR camera	45	24	12 d x 24
Multi-spectral TV camera	110	200	6 x 18 x 24
Radiometer-scatterometer electronics	90	150	12 x 12 x 15
Passive microwave scanner electronics	50	175	12 x 12 x 12
Microwave radiometer electronics	57	80	12 x 12 x 12
Data collection unit	11	8	18 x 18 x 6
Sferics detector	20	6	19 x 17 x 11
Antenna	5	-	Exten. wire
Cloud camera	14	15	9 x 9 x 10
Absorption spectrometer	30	15	20 x 20 x 24
Spectral polarimeter	40	20	8 x 20 x 24
Electronic imaging camera	10	21	6 x 6 x 14
Aeronomy spectrometers			
Michelson	35	-	24 x 15 d
Ebert	50	-	24 x 15 d
Electronics	16	260	9 x 16 x 19
ANTENNA ASSEMBLIES			
Microwave radiometer	70	-	360 d
Radiometer-scatterometer	70	-	360 d
Passive microwave scanner	200	-	
LF-VLF dipole	60		
LF-VLF loop			

Table 5.3-3. Earth Observations Platform Weight Statement

Item	Subtotal	Total
Common Support Module		2179
Structure	690	
Subsystems	1489	
Mission Equipment Modules		6317
Structures	2200	
Docking mechanism (2)	200	
Thermal protection	195	
Extension booms	460	
Telescope assembly	1000	
Replaceable units	1852	
Antennas	410	
Total		8496

Table 5.3-4. Solar Astronomy Sensor Functional Summary

Item	Spectral Range - Å				Measurement Function
	Infrared $10^6 - 10^4$	Visible 8000 - 4000	Ultraviolet 4000 - 40	X-Ray 200 - 0.1	
Proportional counter			100	---- 2	Energy distribution
Magnetograph				0.1	Field strength (~1 gauss)
White-light camera		6000 - 4000			Visible spectrum band
Hydrogen-alpha camera		6560			HOC deep red
Imaging camera		7000 - 3000			Target image
Coronagraph		10^4 - 4000			Plasma clouds (1-6/5-30 solar radii)
XUV spectroheliograph			650 - 170		Solar disc
X-ray spectroheliograph	11,000 — —	— 1300			Fine granulation structure and spectra
Echelle spectroheliograph	11,000 — —	— 1300			
Photoheliograph	11,000 — —	— 1300			Granular structure
X-ray telescope			100	---- 2	X-ray spectra
Crystal spectrometer			20 - 10		X-ray spectra
IR spectrometer	25,000 -	4000			IR radiation
EUV spectrometer			1700 - 30		EUV observations

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Table 5.3-5. Solar Astronomy Platform Equipment Characteristics

Equipment	Weight (lb)	Power (watts)	Dimensions (in.)	Ring Location
X-RAY ASTRONOMY				
0.5 m X-ray telescope	80	40	20 d x 125	-
X-ray spectrometer	75	150	12 x 12 x 24	FWD
Proportional counter	20	22	20 x 20 x 24	FWD
TV field camera and pallet	35	39	5 d x 15	FWD
ASPECT (SPOTTING) SENSOR	160	19	10 d x 84	-
SUN SENSORS				
X-ray spectroheliograph	64	47	35 x 25 x 22	-
X-ray crystal spectrometer	28	40	12 x 15 x 30	FWD
IR scanning spectrometer	65	50	26 x 22 x 13	FWD
EUV spectrometer	25	-	12 d x 10	FWD
PHOTOHELIOGRAPH				
1.5 m telescope	2600	40	60 d x 194	-
Doppler Zeeman magnetograph	200	200	20 x 40 x 24	FWD
White-light TV camera	12	30	6 x 6 x 14	FWD
Hydrogen-alpha TV camera	12	30	6 x 6 x 14	FWD
Echelle spectrograph	90	215	24 x 24 x 24	FWD
Light beam splitters	40	-	10 x 24 x 20	FWD
SOLAR CORONOGRAPH				
Telescopes	595	20	39 x 63 x 130	-
Electrical imaging cameras (2)	20	42	6 x 12 x 14	FWD
XUV SPECTROHELIOGRAPH				
0.25 m telescope	170	20	15 x 52 x 27	-
UV spectrometer	200	50	20 x 20 x 24	MID
SUPPORTING ELECTRONICS				
EUV spectrometer	14	16	9 x 9 x 16	MID
X-ray crystal spectrometer	30	48	12 x 6 x 6	MID
White-light TV	44	70	19 x 14 x 14	MID
Hydrogen-alpha TV	44	70	19 x 14 x 14	MID

Table 5.3-6. Solar Astronomy Platform Weight Statement

Item	Subtotal	Total
Common Support Module		2129
Structure	540	
Subsystems	1489	
Docking Mechanism	100	
Mission Equipment Modules		6172
Structures	1400	
Thermal Protection	119	
Telescope Assemblies	3605	
Replaceable Units	1058	
Total		8301

STELLAR ASTRONOMY GEOSYNCHRONOUS PLATFORM

The stellar astronomy platform is similar in appearance to the solar astronomy grouping but with major emphasis on ultraviolet and X-ray investigations. The wide range of sensors permits several concurrent investigations removed from lower earth altitude influences.

Functional Performance

The selected telescopes, sensors and imaging devices are described extensively in previously referenced NASA source data. In general, the X-ray instrument grouping provides capabilities to measure flux and energy distribution characteristics from selected stellar sources. The ultraviolet grouping will provide for imaging of galactic and nebulae emissions and galaxies or cluster stars. Functional characteristics of various equipments are contained in Table 5.3-7.

Physical Description

The stellar astronomy geosynchronous platform (reference Figure 5.3-3) consists of two standard equipment rings, and a semi-ring structure, to which are mounted the telescope assemblies. The aft (-Z) ring is a common support module with the platform subsystems installed, and a servicing system interface docking ring. The forward ring contains direct-viewing sensors which look forward below the partial ring, and supporting electronics units. The partial ring contains the beam-directing assemblies, telescope sensors, and provides mounting for the two large telescopes and other optics.

Table 5.3-8 defines the characteristics of the mission equipment. An overall weight summary for the platform total of 5896 pounds is contained in Table 5.3-9.

Table 5.3-7. Stellar Astronomy Sensor Functional Summary

Item	Spectral Range, Å					Measurement Function
	Infrared 10^6-10^4	Visible 8000-4000	Ultraviolet 4000-40	X-Ray 200-0.1	Gamma Ray	
UV telescope			4000-900			UV energy gathering and imaging
UV field camera			1300-1000			
UV spectrometer			4000-900			Far UV colors and dimensions
X-ray telescope				100 - 2		(0.1 - 10 Kev)
X-ray camera				100 - 2		Image intensifier
UV TV camera			3000-2000			Broadband UV imaging
Gamma ray spectrometer					.01-.001	(0.06-10 Mev)
Spectrom/polarimeter				5 - 1.5		(6 - 8.3 Kev)
X-ray crystal spectrom.				10 - 1.5		Moderate strength spectra
IR interferometer	160K-50K					Radiation detection

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Table 5.3-8. Stellar/X-Ray Astronomy Platform Mission Equipment

Item	Weight (lb)	Power (watts)	Dimensions (in.)
NARROW-FIELD UV ASTRONOMY			
1.0 m telescope and tracker assembly	858	40	40 d x 145
TV field cameras (4)	120	136	20 x 5 x 15
UV spectrometer	276	50	40 x 30 x 20
X-RAY ASTRONOMY			
0.7 m telescope assembly	320	50	24 d x 150
X-ray image intensifier	70	42	5 x 5 x 24
Transmission grating	20	-	24 d x 4
Field camera	30	34	5 d x 15
ASPECT (SPOTTING) SENSOR	160	19	10 d x 90
MISSION EQUIPMENT RING			
UV TV camera	12	30	6 x 6 x 14
Support electronics	44	70	19 x 14 x 14
Gamma-ray spectrometer	260	5	20 d x 20
Support electronics	20	15	12 x 12 x 12
Spectrometer polarimeter	60	6	20 d x 24
X-ray crystal spectrometer	28	40	12 x 15 x 24
Support electronics	30	48	12 x 15 x 24
X-ray photoelectric polarimeter	20	15	12 d x 18
IR interferometer	40	50	12 x 18 x 24
X-ray image intensifier electronics	40	80	19 x 12 x 24

Table 5.3-9. Stellar X-Ray Astronomy Platform Weight Statement

Item	Subtotal	Total
Common Support Module		2129
Structure	540	
Subsystems	1489	
Docking mechanism	100	
Mission Equipment Modules		3767
Structures	1300	
Thermal protection	59	
Telescope assemblies	1338	
Replaceable units	1070	
Totals		5896

PLASMA PHYSICS GEOSYNCHRONOUS PLATFORM

Several particle and wave payload candidates have been combined onto one platform to provide for a wide range of data gathering. The basic structural elements of the platform are smaller than the previously described platforms, but a number of instruments are extended on booms to provide separation from the platform.

Functional Performance

A number of instruments is included on the platform to meet overall objectives of space physics research. This includes wave particle experiments, cometary, atmospheric, and magnetospheric investigations, and meteoroid science. Table 5.3-10 defines functional performance of the platform sensors.

Table 5.3-10. Plasma Physics Platform Instrument Functions

Item	Functional Characteristics
Mass spectrometer	1 to 8 and 7 to 56 AMU mass 0.1 to 1.6 Kev
Ion temperature/ density monitor	5 to 10^6 e or I/cm^3 density, 500 to 20,000K temperature
Closed quadrupole mass spectrometer	10^8 dynamic range, 1 to 46 AMU mass
LF/VLF transceiver	500 Hz to 100 kHz frequency
Particle detector	0 to 2 Kev, 0.5 to 20 Kev and 10 to 500 Kev particles
Electron accelerator	0.1 to 5 amps, 10 to 20 Kev range
Hemispherical anal.	10^9 to 10^2 e/cm^2/sec
Interferometer/ spectrometer	10^6 to 10^4 angstroms (1 to 100 microns) spectral range
Scanning/grating spectrometer	10^4 to 10^3 angstroms (0.1 to 1.0 micron) spectral range
EUV spectrometer	1700 to 30 angstroms (0.7 to 0.003 micron) spectral range
Cosmic dust sensor	$< 10^{-15}$ kg mass sensitivity

Physical Description

The plasma physics geosynchronous platform (reference Figure 5.3-4) is a very compact configuration, consisting of two equipment rings, some forward (+Z) mounted sensors, and five extendable booms (not including solar array booms). The aft (-Z) ring is a common support module, housing the subsystems, and with an attached docking assembly for servicing. The forward ring houses

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the exterior viewing sensors and the supporting electronics. Larger sensors extend forward of this ring. Three sensors and two antennas extend outboard after deployment on structural astrobooms. All of the mission equipment characteristics are identified in Table 5.3-11. The total platform assembly has a gross weight of just over 4100 pounds. The weight statement is summarized in Table 5.3-12.

Table 5.3-11. Plasma Physics Platform Mission Equipment

Item	Weight (lb)	Power (watts)	Dimension (in.)
FORWARD RING SENSORS			
Mass spectrometer	8	-	6 d x 10
Ion/electron monitor	13	12	10 d x 6
Quadrupole mass spectrometer	18	18	10 d x 7
Electron accelerator	100	130	70 d x 12
Interferometer/spectrometer	166	-	24 x 20 x 20
Scanning/grating spectrometer	186	-	24 x 40 x 20
EUV spectrometer	25	-	12 d x 10
Cosmic dust sensor	10	-	16 x 12 x 2
SUPPORT ELECTRONICS RING			
Mass spectrometer	23	35	8 x 8 x 4
LF-VLF transceiver	27	70	19 x 19 x 15
Interferometer/spectrometer	68	457	19 x 21 x 16
Scanning/grating spectrometer	68	457	19 x 21 x 16
EUV spectrometer	14	16	9 x 9 x 16
Cosmic dust sensor	14	10	19 x 14 x 6
EXTERNAL INSTALLATIONS			
LF-VLF antennas	70	-	1476 tip
Magnetometer-fluxgate	23	20	10 x 10 x 10
Hemispherical analyzer	25	10	20 d x 2
Particle detector	75	50	12 x 20 x 24

Table 5.3-12. Plasma Physics Platform Weight Statement

Item	Subtotal	Total
Common Support Module		2059
Structure	470	
Subsystems	1489	
Docking mechanism	100	
Mission Equipment Modules		2043
Structure	800	
Thermal protection	80	
Replaceable units	740	
External installations	423	
Totals		4102

HIGH-ENERGY PHYSICS GEOSYNCHRONOUS PLATFORM

The high-energy physics platform contains a grouping of mission equipment for gamma and cosmic ray energy detection functions in investigative areas such as nucleonic anti-matter and extra-heavy nuclei. The various equipment items support a range of experimental objectives in cosmic ray physics.

Functional Performance

The selected instruments scaled to the characteristics of geosynchronous orbit are described fully in the previously referenced NASA source data. Typical functional capabilities of the primary mission equipment are summarized in Table 5.3-13.

Table 5.3-13. High-Energy Physics Functional Performance

Item	Function
Total absorption shower counter	Electron energy measurement
Cerenkov counter	Electron energy measurement
High Z detector	Nucleon energy measurement
Magnetic spectrometer	Particle kinetic energy, 200 to 500 GV
Gamma ray detector	100 Kev to 10 Mev energy range
Cosmic ray detector	7 Mev to 300 Mev energy range

Physical Description

The high-energy physics geosynchronous platform (reference Figure 5.3-5) consists of one common support module ring and an extended equipment bay containing very large and heavy sensor equipment. The servicing docking ring is attached to the common support module which houses the subsystem equipment. Because of the unique viewing needs of the mission equipment, a ring arrangement for the mission equipment was not feasible. Servicing therefore could require some difference in manipulator movement. The overall configuration is characterized by the length of the cylindrical section and by the "windows" for sensor viewing. Table 5.3-14 identifies the characteristics of the on-board mission equipment. The platform has an overall gross weight of 8499 pounds. A summary of the weight statement is contained in Table 5.3-15.

Table 5.3-14. High-Energy Magnetospheric Physics Platform

Item	Weight (lb)	Power (watts)	Dimensions (in.)
Total absorption shower counter	3000	44	40 x 80 x 30
Cerenkov counter	1020	10	40 x 40 x 36
High-Z detection windows	304	-	40 x 40 x 6
Gamma-ray detector	36	10	8 d x 12
Cosmic ray detector	130	15	30 d x 48
Magnetic spectrometer	200	500	66 x 30 x 30

Table 5.3-15. High-Energy Physics Platform Weight Statement

Item	Subtotal	Total
Common Support Module		2129
Structure	540	
Subsystems	1489	
Docking mechanism	100	
Mission Equipment Module		6370
Structures	1600	
Thermal protection	80	
Fixed assemblies	3220	
Replaceable assemblies	1470	
Totals		8499

5.4 OBSERVATIONAL PLATFORMS IMPLEMENTATION PLAN

The implementation plan for the observational platforms is based upon replacing the satellites listed in the traffic models with a platform at the first scheduled launch of a new satellite during the space shuttle era.

Figure 5.4-1 illustrates the approach for the baseline traffic model. No specific designation is assigned to the astro-physics platforms. They just correspond to the four launches listed in the baseline traffic model. Earth observational platforms include both the meteorological and earth resource satellite functions. Note also that the on-orbit servicing concept precludes the necessity for replacement every three to four years. The space element inventory is reduced from 19 to only 4.

Figure 5.4-2 presents the implementation plan for observation platforms that correspond to the requirements of the new traffic model. Because the traffic model essentially reflects the astro-physics mission equipment definition derived for the baseline traffic model, the only difference in the satellite and platform plans is the satellites are replaced, whereas the platforms can be serviced and updated. As a result of the on-orbit servicing concept the earth observation space element inventory is reduced from 10 satellites to four platforms. The astro-physics inventory reduction is less significant because of the assumption that foreign countries are building to the basic complement of four astro-physics platforms/satellites. Therefore, only two satellite replacements by Russia (Region II) and the U.S. (Region IV) are negated by the platform concept.

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	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990
ASTRO-PHYSICS (REGION IV)		△ ▲			△ ▲			△ ▲			△ ▲
EARTH OBSERVATIONS											
SYNCH. METEOROLOGICAL											
REGION I		△				△				△	
REGION II				△				△			
REGION III			△				△				△
REGION IV	△				△				△		
SYNCH. EARTH RESOURCES											
REGION I						△			△		
REGION II						△			△		
REGION III						△			△		
REGION IV						△			△		
EARTH ORBITAL PLATFORMS											
REGION I	▲										
REGION II				▲							
REGION III			▲								
REGION IV		▲									

LEGEND: △ SATELLITE DELIVERIES
▲ PLATFORM DELIVERIES

Figure 5.4-1. Baseline Traffic Model Observation Platforms Implementation Plan

SD 73-SA-0036-5

Space Division
North American Rockwell

	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990
EARTH OBSERVATION											
REGION I			△ ▲			△					
REGION II	△ ▲										
REGION III	△ ▲			△			△				
REGION IV			△ ▲			△					
ASTRO-PHYSICS											
REGION I					△ ▲				△ ▲		
REGION II	△ ▲		△ ▲		△ ▲		△ ▲		△		△
REGION III	△ ▲										△ ▲
REGION IV	△ ▲		△ ▲		△ ▲		△ ▲		△		△

LEGEND:

△ SATELLITE DELIVERIES

▲ PLATFORM DELIVERIES

Figure 5.4-2. New Traffic Model Observation Platforms Implementation Plan

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